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SHARC

Space Habitat, Assembly and Repair Center

Submitted to:
Dr. George Botbyl
The University of Texas at Austin
Department of Aerospace Engineering

Submitted by:
Integrated Space Systems

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Todd Colangelo, Team Leader

Philip Tran

Debora Hoetger

Chris Tutt, Chief Administrator

Addison Kuo

Chad Wassmuth

Michael Lo

Gregory Wildgrube

Leland Marcus, Chief Engineer

Executive Overview

Integrated Space Systems (ISS) has taken on the task of designing a Space Habitat, Assembly and Repair Center (SHARC) in Low Earth Orbit to meet the future needs of the space program. Our goal is to meet the general requirements given by the 1991/1992 AIAA/LORAL Team Space Design competition with an emphasis on minimizing the costs of such a design. This semester, we have created a baseline structural configuration along with preliminary designs of the major subsystems.

Assumptions and Requirements

Our initial mission requirements, which were set by AIAA, were that the facility be able to:

- Support simultaneous assembly of three major vehicles
- Conduct assembly operations with minimal EVA
- Maintain orbit indefinitely
- Assemble components 30′ long with a 10′ diameter in a shirt-sleeve environment

Our group also made several assumptions to further refine the mission parameters:

- "Three major vehicles" were defined as two lunar vehicles and one Mars vehicle. For relative sizes, see Table A.
- SHARC must begin limited operations after eight launches.
- No HLLV of Shuttle-C will be available.
- The maximum crew size is eight and the maximum work tour is 35 days.
- A garbage collection system will be available to deal with orbital debris.

With these assumptions in mind, we began conceptual designs of SHARC's baseline configuration.

Table A: Interplanetary Vehicle Sizes

Vehicle	Total Mass	Fuel Mass	Max. Dia.	Length
PhTV	1311.3 mt	811.5 mt	23.1 m	58.4 m
PhCV	467.0 mt	262.8 mt	18.8 m	43.1 m
MTV	Not given	Not given	27.4 m	8.3 m
LTV	94.1 mt	80.9 mt	13.7 m	6.9 m
LTS	191.7 mt	159.2 mt	15.2 m	22.9 m

Structural Configuration

Twelve different conceptual designs were reviewed (see Appendix B) using a decision matrix. The designs we looked at were versatile enough to accommodate most of the different subsystem concepts we considered. Our chosen design is called the Hammerhead II.

The Hammerhead II configuration, shown in Figure A, will be composed of two 35′ x 200′ double deployable trusses separated by four 35′ erectable trusses. There are two smaller bays for lunar vehicles and one large bay for assembling the Phobos and Mars Transfer vehicles. A track system mounted by remote manipulator arms will encircle each bay allowing the arms to assist in vehicle assembly, hence minimizing EVA. There will be a total of seven robotic arms to help in vehicle assembly: one 30 ft arm for each lunar bay, two 30 ft arms for the Mars bay, one 30 ft arm for storage of parts, and two 60 ft arms located on the sides of the main deployable trusses for berthing and transporting payloads.

A general storage area is located in the 21′x50′x35′ area between the two double fold deployable trusses, making it easily accessible to all assembly bays. An alternate storage area is located on the double fold deployable truss leading out to the solar arrays, which is accessible by a robotic arm. The spring-loaded 31′x14′ diameter Phobos fuel tanks will be located near the Mars bay ready to be jettisoned for safety.

The emergency escape pod will be located in the center of the four habitation and control modules and will be accessible from two pressurize corridors for quick use. The modules are arranged in a racetrack configuration to provide dual egress in case of emergencies. The two control modules will contain windows which will overlook the lunar bays to help in vehicle assembly and payload berthing.

The eight sets of solar arrays and the battery system are located at the end of the double fold deployable truss. The 40x20 ft pressurized sleeve, which is attached to the airlock, can contain a 30x10 ft component and is accessible to the robotic arms. Finally, the shuttle will dock upside down to the remaining airlock. This provides plenty of clearance for docking, and the Shuttle can be

rigidly connected to the double fold deployable truss through attachment points in the Shuttle payload bay.

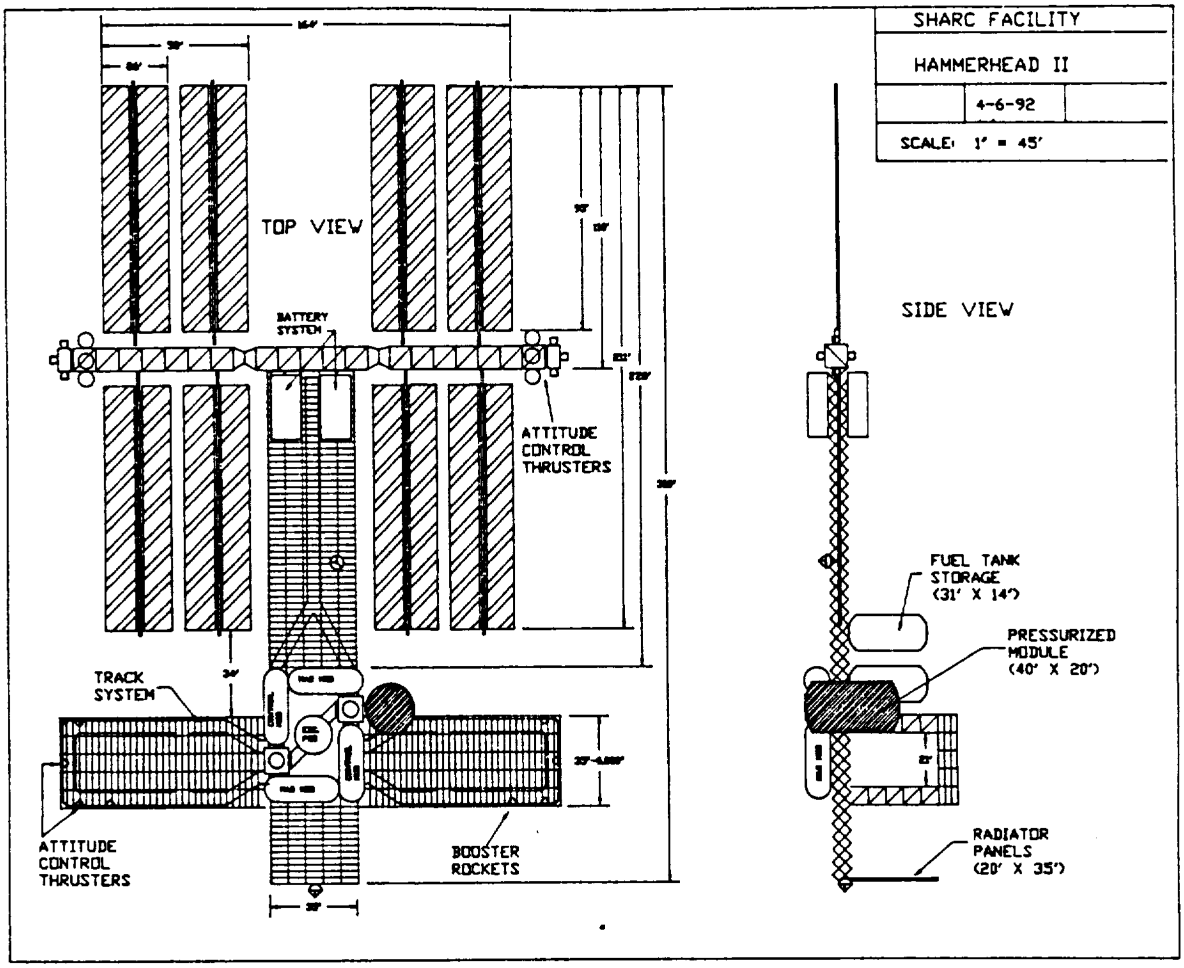

Figure A Hammerhead II Configuration

Orbit and Altitude

We determined that the orbit of SHARC should be at an inclination of 28.5^{o} and altitude of 380 km. This altitude is accessible to all current medium and heavy lift launch vehicles in use with only minor reductions in payload capacity. The inclination angle was chosen because it provides an ideal transportation node for future Mars and Lunar exploration missions. This inclination can also be reached by rockets from both the Kennedy Space Center and Kourou. We determined the Ballistic numbers of SHARC using a simplified model and then recalculated the results using a much more accurate model. We also considered utilizing the Space Station Freedom as a habitation depot for the workers at SHARC but calculations showed that the synodic period of the two facilities was 14.5 days.

Crew and Life Support

A work tour on SHARC will consist of a maximum crew of eight over a period of 35 days. The shuttle will stay docked at SHARC for the full duration of the mission. Each astronaut would work for 8 hours per day, 6 days a week. Life support supplies would be carried on the Shuttle, with any assembly materials carried on an unmanned vehicle which would be launched from 3 to 10 days after the Shuttle.

The Crew and Life Support group performed sizing estimates for a closed-loop life support system involving full air and water recycling. Further calculations were made involving specific supply requirements. Preliminary estimates reveal that 147 kg of nitrogen gas and 343 kg of food will be required for each work tour. 107 kg of methane and 183 kg of solid waste matter will be generated during the work tour and will have to be removed.

Power

The amount of power required to run SHARC was determined by compiling the amount of power required by each subsystem, along with estimated values for special items such as exterior flood-lighting for bays, robotics, power tools, and EVA. This method resulted in a power requirement of 62 kilowatts. Assuming 10% line losses, the total power required was 68 kW.

Photovoltaic silicon solar arrays were chosen as the primary power system. From several calculations it was determined that a total area of 1854 m^2 was required to provide the 68 kW of power. The arrays are arranged as eight pairs of fold-out panels which deploy along an erectable mast or boom for stability. The total mass of the arrays is 2267 kg and have a calculated lifetime of 10 years after which they will have experienced approximately 25% degradation in efficiency.

The storage system chosen to power SHARC during eclipse periods were 27 Nickel-Hydrogen ($Ni\text{-}H_2$) individual pressure-vessel batteries connected in parallel for increased capacity and redundancy. The batteries are arranged together in groups and are placed in thermally controlled cases for optimum performance. The cases are placed between the two large sets of solar arrays. Each $Ni\text{-}H_2$ battery has a capacity of 100 amp-hours, an energy density of 25 W-hr/kg, and a mass of 112 kg. The total mass of the battery system (not including wiring) is approximately 3024 kg. For a worst case scenario, the batteries have a lifetime of 2 years if they are required to generate continuous peak power. Using a more probable average power of 48 kW, the lifetime increases to 5 to 6 years. After this time, the batteries will experience significant degrading and must be replaced.

Robotics

The construction and operation of SHARC will require the extensive use of robotics. The need for robotics stems from the hazardous nature that long-term EVA operations would present to astronauts and the need to relieve crew work loads. In addition, SHARC's main purpose of servicing space vehicles necessitates the use of robotics.

Two principal robotic systems were selected for use on SHARC: a remote manipulator system (RMS) and flight telerobotic servicer (FTS). These two systems are advanced versions of the ones to be used on Space Station Freedom. The use of robotic systems like these would reduce the uncertainties and costs in building SHARC.

On SHARC, the primary function of the RMS will be to capture and move large cargo and parts of spacecraft to be assembled around the service area. Then the FTS will attach itself, or be transported by the RMS, to the work site and proceed to work on light, precision assembly tasks. The FTS will also be able to examine the structural elements of SHARC for maintenance purposes.

GNC/ Reboost

The GNC/Reboost subsystem determined the propulsion requirements of SHARC during operation in space. Based upon our drag model, the propulsion system must be able to reboost SHARC from an altitude of 364 km to 380 km every two months. The total required ΔV was found to be 9.107 m/s. In addition, SHARC will be rotated 90 degrees during reboost periods, and there will be enough propellant stored to allow one additional reboost without re-supply. The location of the attitude thrusters and the reboost thrusters is shown in Figure A.

Propellants were compared on the basis of specific impulse and storage requirements. Hydrazine (N_2H_4) was selected for standard attitude control, while the reboost thrusters will use an OME/UR bipropellant (N_2O_4/MMH) rocket produced by Aerojet.

Communications

The communications subgroup used existing SSF information as a basis for choosing the communication system for SHARC. Communications will be separated into a local system and a space to ground system. The local system will consist of an optical network because of its low power requirements and higher efficiency. The maximum data rate for the local system is 10 Mbps (Megabits per second) with the option of using point to point fiber optics for a maximum data rate of 100 Mbps.

The space to ground system will consist of two virtual channels operating at a data rate of 150 Mbps. The frequency will be in the range of approximately two gigahertz to overcome any atmospheric or noise attenuation. The data will be transmitted to the Tracking and Data Relay Satellite System (TDRSS) and then to the Data Interface Facility which will allocate the data to the appropriate users. This link design will maintain continuous contact with the ground stations so that tracking and telemetry can be monitored.

Thermal Control

The first objective of the thermal control group was to identify the different station elements that have specific temperature limits. After these temperature limits were determined, various passive thermal measures were studied to determine if they would be adequate by themselves. This proved true in the case of the cryogenic fuel tanks. For the rest of the station, we estimate that a peak load of 60 kW of waste heat must be dissipated. An active thermal control system was designed using Freon-12 as a working fluid. A radiator panel 35′ x 20′ was found to be adequate for our needs.

List of Acronyms

AC	Alternating Current
ACU	Arm Computer Unit
AIAA	American Institute of Aeronautics and Astronautics
C_d	Coefficient of Drag
DC	Direct Current
DIF	Data Interface Facility
DOF	Degree of Freedom
ΔV	Delta V
EC/LS	Environmental Control/Life Support
EPS	Electrical Power Subsystem
EVA	Extra Vehicular Activity
FCC	Federal Communication Commission
FTS	Flight Telerobotic Servicer
GN&C	Guidance, Navigation, and Control
ISS	Integrated Space Systems
JSC	Johnson Space Center
LEE	Latch End Effector
MMH	Monomethyl Hydrazine
NASA	National Aeronautics and Space Administration
PBS	Payload Berthing System
RMS	Remote Manipulator System
SAMSIN	Servo-Actuated Manipulator System with Intelligence Networks
SHARC	Space Habitat, Assembly and Repair Center
SSF	Space Station Freedom
SEE	Standard End Effector
TOF	Time of Flight
TDRSS	Tracking and Data Relay Satellite System
UST	Universal Servicing Tool

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1.0 General Summary

1.1 Project Overview

The Space Habitat, Assembly and Repair Center (SHARC) is a design of a Low Earth Orbit Assembly Facility, and is our entry in the 1992 AIAA/LORAL Team Space Design Competition. It is capable of supporting the interplanetary missions described in the Office of Exploration FY1988 and FY1989 reports for the mission timelines presented with those mission. The projected fabrication schedule estimates that six launches will be needed to assemble enough of the station components to begin basic operations. This would occur in 1998 and full operational status would be achieved in 2005.

1.2 Orbital Parameters

Selection of an orbit was driven by two factors: SHARC's designed mission as an interplanetary transportation node and orbital decay due to atmospheric drag. Orbital inclination was set at 28.5°. This is accessible from both the Kennedy Space Center and Kourou. It is also the average inclination of the Moon's orbit, allowing departing lunar vehicles to avoid the fuel penalties associated with plane changes. The nominal altitude of SHARC is 380 km, well within the operational capabilities of both the Shuttle and the current medium and heavy unmanned launch vehicles in use. At this altitude, the station's orbit decays 20 km every 60 days, the interval between shuttle visits. ΔV's for a Hohmann transfer were calculated and fuel requirements were found to be reasonable.

SHARC is maintained in a local vertical altitude, with the solar arrays mounted on gimbals to allow them to face the sun as much as possible during the orbit. This attitude allows us to take advantage of the gravity gradient for orbital stability. Since SHARC does not require precise pointing accuracy, this effect can provide the bulk of attitude control.

1.3 Station Configuration

After several conceptual design iterations, the Hammerhead II configuration was selected by our group as the primary design for the station. A CAD drawing of the station is given in Figure 1.3.1. Several unique features were incorporated in the design to accommodate the AIAA mission requirements. A system of tracks covers most of the station along which several telerobotic arms move. These arms allow assembly work to proceed without requiring extended EVA. The cryopropellants are stored in large tanks next to the bay areas, where the arms have easy access to them. Assembly materials are stored in the large central bay, where again the arms can easily move them to wherever they are needed. The pressurized modules are aligned along the top of the storage bay in a racetrack configuration, allowing the astronauts dual egress in case of catastrophic failure of one of the modules.

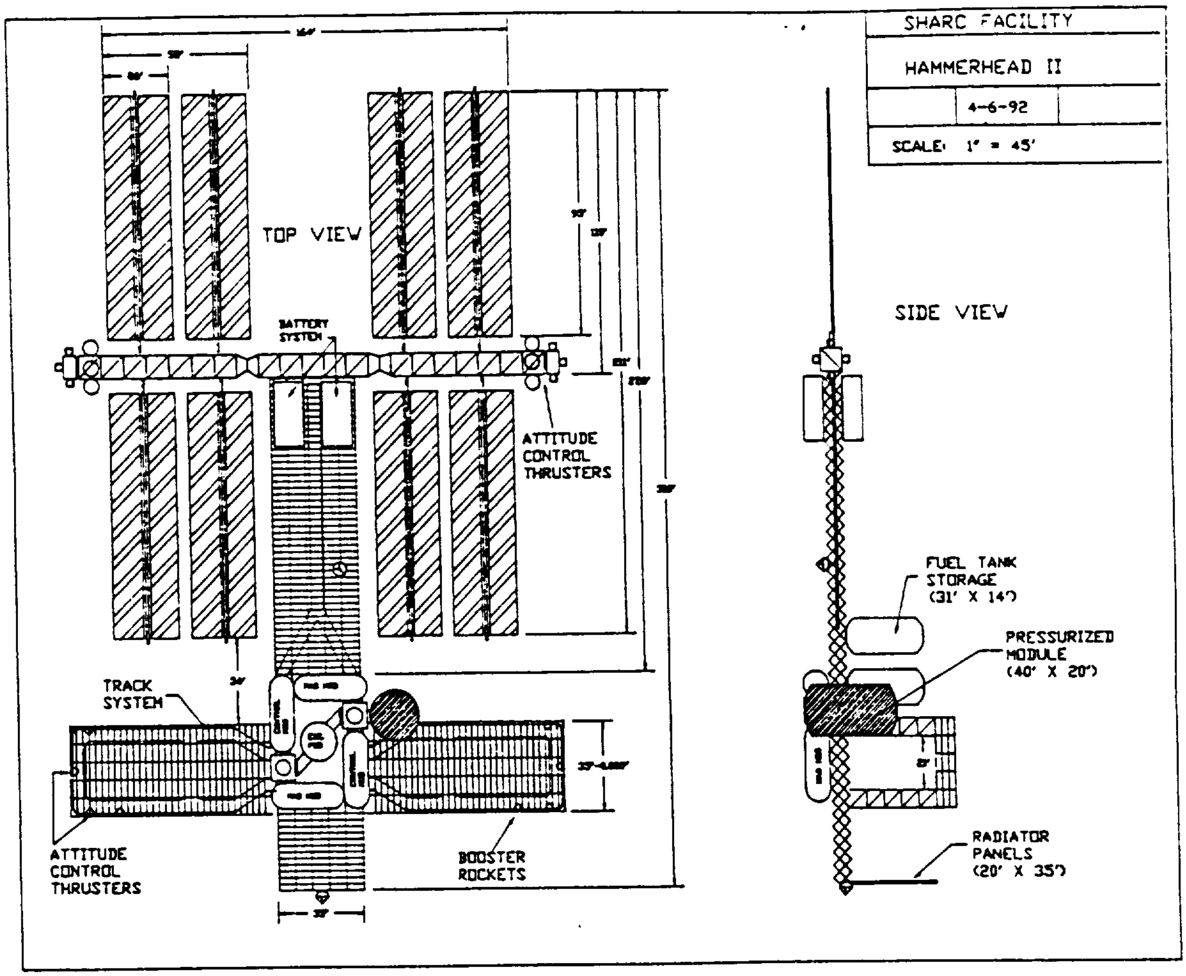

Figure 1.3.1 Hammerhead II Configuration

1.4 Other Subsystems

1.4.1 Environmental Control and Life Support

The original crew scenario for SHARC relied heavily on interaction with Space Station Freedom. After some analysis, however, this did not prove feasible. Our revised crew scenario called for a maximum crew of eight to stay on SHARC for 35 days. Based on this work tour, we sized a closed-loop life support system, with full air and water recycling. From this, we determined the supplies required for an average work tour and the amount and type of waste products generated.

1.4.2 Power Supply

Our power group looked at several possible types of power generation systems for SHARC. After analyzing the options, we selected photovoltaic solar panels as our primary power source with Ni-H_2 batteries as our backup for when the station is in the Earth's shadow. Based on a subsystem "power budget," which estimated peak loading conditions for each of the major subsystems, a size estimate for the primary and secondary systems was created.

1.4.3 Communications

Two widely variant communication systems are required on SHARC. Because assembly payloads will be delievered by unmanned vehicles, one system is needed to control this rendezvous and docking operation. A second system is required to maintain constant audio/visual contact with JSC. An optical system was chosen for the first system and a standard antenna system, which was sized and integrated into the station configuration, was chosen for the second. In

addition, work was done on maintaining contact with the telerobotic equipment used in the station.

1.4.4. Thermal Control

Our thermal group came up with an estimate for waste heat generated or absorbed by the station. Using an energy balance method, we determined how much heat would have to be removed from the modules to return them to a habitable temperature. Based on this, we sized an active thermal control system using a radiator panel. Our group also looked at thermal control of the cryopropellant tanks, determining that their designed passive control system was adequate for our needs.

2.0 SHARC Configurations and Timeline

2.1 Mission Requirements and Assumptions

The AIAA/LORAL contest RFP has very specific mission requirements. At a minimum, SHARC must be able to:

- support simultaneous assembly of three major vehicles
- store vehicle parts and allow easy access from the assembly bays
- minimize EVA with robotic and teleoperated assembly systems
- maintain orbit indefinitely
- assemble parts up to 30' long (10' dia.) in a "shirt-sleeve" pressurized environment
- receive payloads from a variety of international launch vehicles

Our group added an additional requirement:

- SHARC must begin limited operations after the eighth launch

One of our first tasks was to make assumptions about the technology available for SHARC. We decided early on to use off-the-shelf technology as much as possible. With this idea in mind, we made the following assumptions:

- No HLLV or Shuttle-C will be available. SHARC must be built using existing launch technology.
- "Three major vehicles" were defined as one Mars vehicle and two lunar vehicles.
- The maximum work crew will be eight astronauts.

- Space Station Freedom will be operational, but not needed.
- A general emergency escape pod will already have been designed.
- A garbage collection system will be operational to collect or deflect orbital debris.

With these assumptions, we began looking at overall station configurations. The primary and secondary design were chosen, using a decision matrix, from twelve conceptual designs. The primary design is the Hammerhead II and the secondary is the Hammerhead.

2.2 Hammerhead II

The Hammerhead II design has three open assembly bays for the Lunar and Mars vehicle and one enclosed storage bay. The main structure is two 35' by 200' double deployable trusses, separated by four 15' erectable trusses. The fuel tanks are located half way along the vertical truss, midway between the pressurized modules and the solar arrays. A racetrack module configuration will be used with the two habitation modules, two control modules, one pressurized sleeve, and one escape pod. Mobile cranes on a track system will be used to transfer payload around the station. The Hammerhead II configuration is shown in Figure 1.3.1.

2.3 Hammerhead

The Hammerhead design has two open assembly bays for the Lunar and Mars vehicle and one enclosed bay. The main structure is constructed of 15' erectable trusses with the secondary structure using 9' deployable trusses. The fuel tanks are located as far as possible from the pressurized modules, being

placed next to the solar arrays. A racetrack module configuration is used with three habitation modules, 3 control modules, and one pressurized sleeve. Mobile cranes on a track system are used to transfer payload around the station. The Hammerhead configuration is shown in Appendix B.

2.4 Startup Operations

The AIAA Request for Proposal listed several interplanetary missions that SHARC must be able to support. Using NASA timelines for the various mission case studies, we created a mission timeline, showing our expectations for SHARC's operational status.

One of the first things that the ISS design team had to determine was a fabrication schedule for SHARC. Based on this schedule, we could estimate the time it would take to deploy and assemble enough of the station for limited operations to begin. The mission required that this occur after the eighth launch. Initial analysis shows that our primary configuration can begin limited operations after six launches. The deployment sequence is shown in Table 2.4.1. (Section 3.0 gives full details on the actual structures involved.)

Table 2.4.1: SHARC Deployment Schedule

Note: Each mission consists of two launches - one manned Shuttle (M) and one unmanned rocket (U)	
Mission 1M	One deployable truss Complete solar array
Mission 1U	One deployable truss Four erectable trusses
Mission 2M	One habitation module
Mission 2U	Main communications antenna Rendezvous communications antenna Initial robots and tracks
Mission 3M	One command module
Mission 3U	Thermal array More robots and tracks
Basic operational capability after six launches	

2.5 Satellite and Probe Capability

After three missions, SHARC will be able to service and assemble itself, speeding up the fabrication process. In addition, the station can begin working on earth-orbiting satellites and interplanetary probes, including robotic precursor mission for lunar and Martian bases. Detailed construction will still have to be done on Earth, however, until the pressurized garage can be deployed. SHARC must be able to support subsystem validation for small payloads. Storage requirements would be minimal.

NASA plans call for robotic precursor missions as early as 2000 for the lunar initiatives (OEXP, p. 2-51, Cohen, p. 4-2, Stafford p. 40). Because of this, we decided to have SHARC launched in 1998. Most of 1998 will be used in assembling the station. In 1999, earth satellite support operations can begin.

This gives enough time to streamline the orbital assembly processes before the precursor missions arrive.

2.6 Phobos Transfer Vehicle Capability

The Phobos Transfer Vehicle described in Case Study 1 of the OEXP FY1988 report (p. 2-13) was designed without an assembly facility like SHARC in place. All orbital assembly tasks are relatively simple things that can be done by the Space Shuttle mechanical arm. By the year 2000, when construction is scheduled to begin, SHARC should be able to dock and store large amounts of incoming payload, especially cryogenic fuel. The robotic assembly equipment, however, does not need to be very sophisticated for this type of work.

2.7 Lunar Transfer Vehicle Capability

Most plans for lunar bases call for construction to begin in 2003 (OEXP, p. 2-79, Stafford p. 40) or 2004 (OEXP p. 2-51, Cohen p. 4-2). In both cases, SHARC must be able to support regular lunar travel. Therefore, we are requiring that lunar vehicle construction begin in 2002. This gives us one year to allow the assembly process for larger vehicles to mature.

Lunar vehicle operations will require in-orbit LOX/LH_2 refueling, subsystem inspections (especially the aerobrakes), and detailed orbital assembly operations.

2.8 Mars Transfer Vehicle Capability

The most difficult task for our station will be the construction of large manned Mars vehicles. Assembly schedules for these mission vary extensively, but the earliest is 2005 (OEXP, p. 2-79). Soon afterwards, SHARC must be able to support regular traffic to and from Mars, including vehicle inspection and refueling.

This is the final mission that SHARC is designed to support. The mission timeline is summarized in Table 2.8.1 below.

Table 2.8.1: SHARC Mission Timeline

1998	SHARC deployment begins
1999	Satellite and probe capability
2000	Phobos Transfer Vehicle capability
2002	Lunar Transfer Vehicle capability
2005	Mars Transfer Vehicle capability

3.0 Structures and Storage

3.1 Overview

The structural subsystem of the SHARC bears the acceleration, thermal, docking loads and provides physical support for various other subsystems. The structure subsystem will cover the structural foundation, material selection, module configuration and attachment, track system, and docking and berthing. The storage subsystem is responsible for the storing of parts and fuel for the station and vehicles.

3.2 Structural Foundation

The space station structural configuration will be built using double fold deployable trusses and erectable trusses made out of 7075 T6 Aluminum. Two 35 ft by 200 ft double fold deployable trusses will be used for the primary construction. They will be carrying all the major loads created by docking, reboost, and robotic operations. Four 35 ft erectable trusses will be used to separate the two double fold deployable trusses. The enclosed bay created will be used as a storage area for parts and tools. The combined weight of the truss structure will be approximately 22,000 lb or 48,000 kg (*Configurations*, p 7). The advantages and disadvantages of each truss type are listed below (*Configurations*, p. 531-534):

- Erectable Truss: The erectable truss has the highest stiffness to density ratio and packing density of all the trusses considered. Packing density is critical because the more truss members that can be packed into the shuttle bay, the less shuttle trips are needed to get the trusses into space. This truss also has the lowest number of elements, making it easier to

maintain. The disadvantages of the erectable truss are that it will take considerable amounts of EVA time to assemble and it lacks redundant elements for a fail-safe structure.

- Double Fold Deployable: The advantages of this truss are its high stiffness to density ratio, high redundancy elements, low EVA time to assemble, and fairly high packing density. The disadvantage is that it has the highest number of truss elements in all the truss configurations we have researched. A picture of the double fold deployable is shown in Figure 3.2.1.

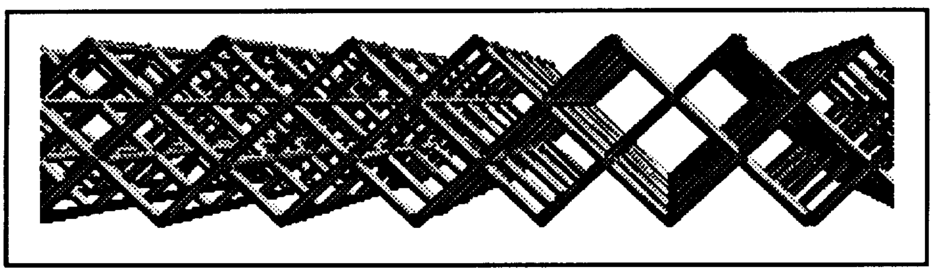

Figure 3.2.1 Double Fold Deployable

A 250 member NASTRAN analysis of the double fold deployable was done to ensure it could withstand the high reboost loads. It was discovered that the truss can withstand the reboost loads in the axial direction but may have problems with the out-of-plane bending and torsion load direction.

3.3 Material

The materials that were researched for the structural configuration are listed below:

- Aluminum
- Steel
- Magnesium
- Beryllium
- Composites
- Kevlar
- Ceramics
- Titanium

The materials used for the construction of the station were selected not only on the basis strength and stiffness, but also on thermal characteristics, corrosion resistance, fracture and fatigue strength, sublimation, electrical and magnetic properties, and ease of manufacturing. Only aluminum, austenitic steel, and titanium met the majority of these criteria. These materials are discussed below (Fraser, et al, pg. 10):

- Aluminum: This material has a high stiffness to density ratio, excellent corrosion-resistance, high ductility, moderate cost, and non-magnetic properties. The disadvantage of aluminum is its low yield strength.

- Titanium: The advantages of this material are the highest stiffness to density ratio and non-magnetic qualities. The disadvantages of titanium are the difficulty in machining the parts and the high cost. Titanium is a good material for low temperature applications, such as cryogenic fuel storage.

- Austenitic Steel: This material should be utilized for high strength regions where titanium is not desirable, due to machining or temperature restrictions.

It was decided that the truss members will be made out of aluminum while the truss members at the high load areas will be made out of steel or titanium.

3.4 Module Configuration

The module configuration of the station has important effects on mission viewing, physical accommodations, and controllability. The configurations that were researched were the cluster, branched, and planar. The planar configuration (racetrack), shown in Figure 3.4.1 was chosen based on the following criteria:

- Provide double egress in case of emergency
- Provide viewing of assembly
- Reduce traffic congestion while traversing modules
- Provide plenty of surface area for thermal controls
- Maximize total work space volume
- Ease of assembly

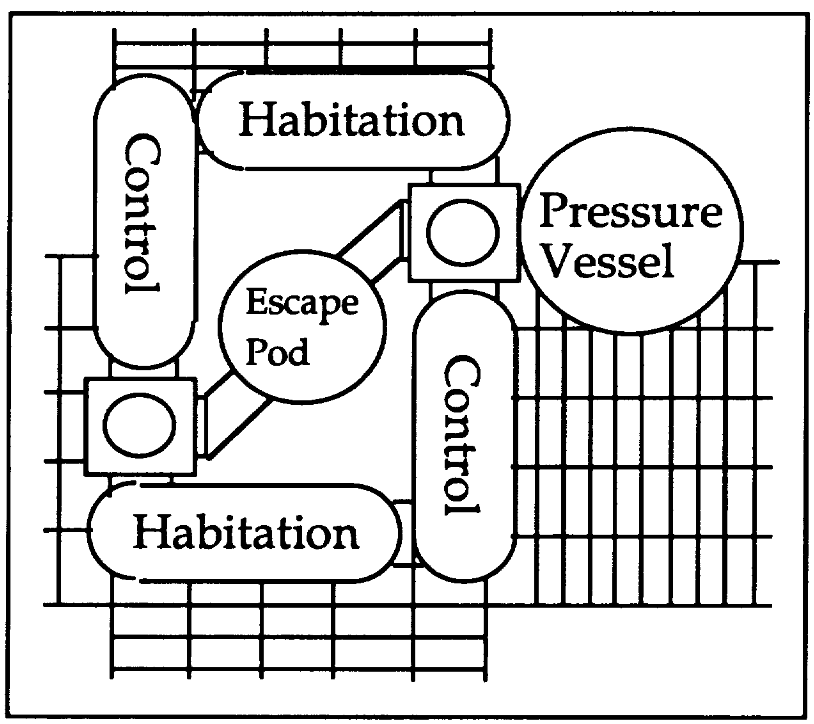

Figure 3.4.1 Racetrack Configuration

Each module will be 30 ft by 10 ft and the airlocks will be 10 ft by 10 ft. In case of an emergency, an escape pod is located at the center of the module configuration to provide transportation for eight crew members to earth. The pod will have two pressurized entrances to ensure quick escape if needed. The 40 ft by 20 ft pressurized sleeve will be attached to one of the two airlocks while the space shuttle will be docked at the other. The pressurized vessel will open up towards the lunar bay, allowing access to a remote manipulator arm.

3.5 Module Attachments

The attachment of the modules and payload to the truss structure needs to be done effectively and efficiently in order to construct a manned orbiting station. Studies have shown that unplanned misalignments between the payload and the truss node geometry will require adjustments by the crew. Special adjustable trunnions will need to be employed to deal with this problem, as shown in Figure 3.5.1 (*Configurations,* p.554).

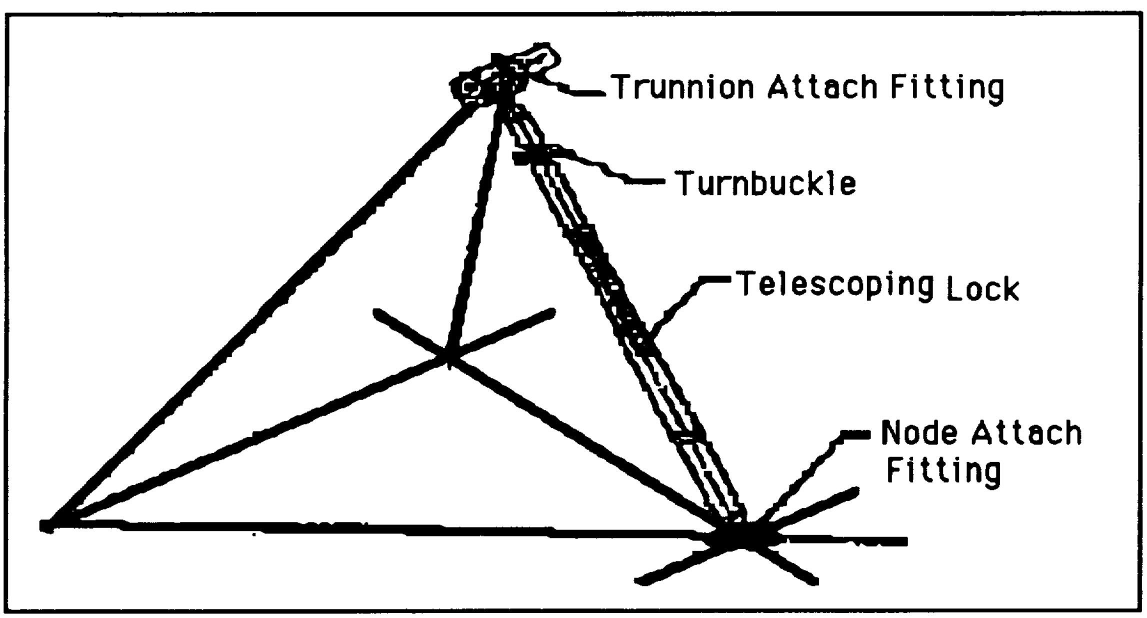

Figure 3.5.1 Adjustable Trunnion

Also, to properly transfer payload loads onto the truss structure, the trunnions need to be connected directly to the node sections of the truss structure. A typical module attachment is shown in Figure 3.5.2 (*Configurations*, p. 561).

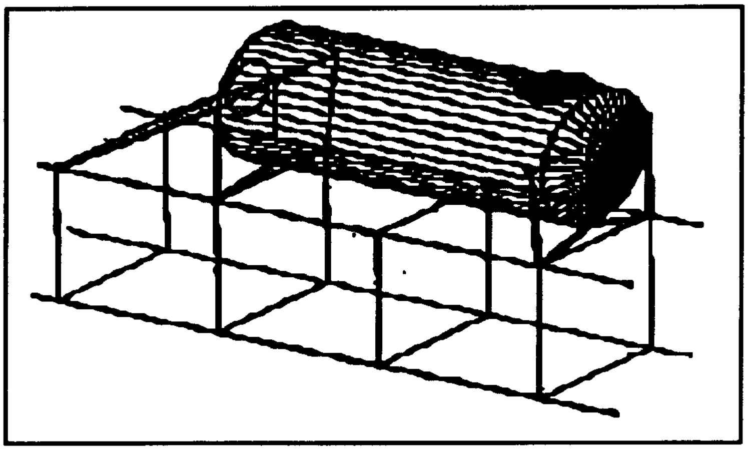

Figure 3.5.2 Module Attachment

3.6 Storage

The main storage will be in the 21ft by 40ft by 35 ft bay, located between the two double fold deployable trusses. This location is easily accessible to all three assembly bays. Alternate storage is located on the double fold deployable leading out to the solar arrays. The storage of the 31 x 14 ft fuel tanks will be

located on the under side of the double deployable truss leading off to the solar arrays. This location keeps the fuel tanks away from the habitation modules but close to the assembly bays. The tanks will be attached in such a way that they can be jettisoned by springs if they get close to dangerous pressure levels.

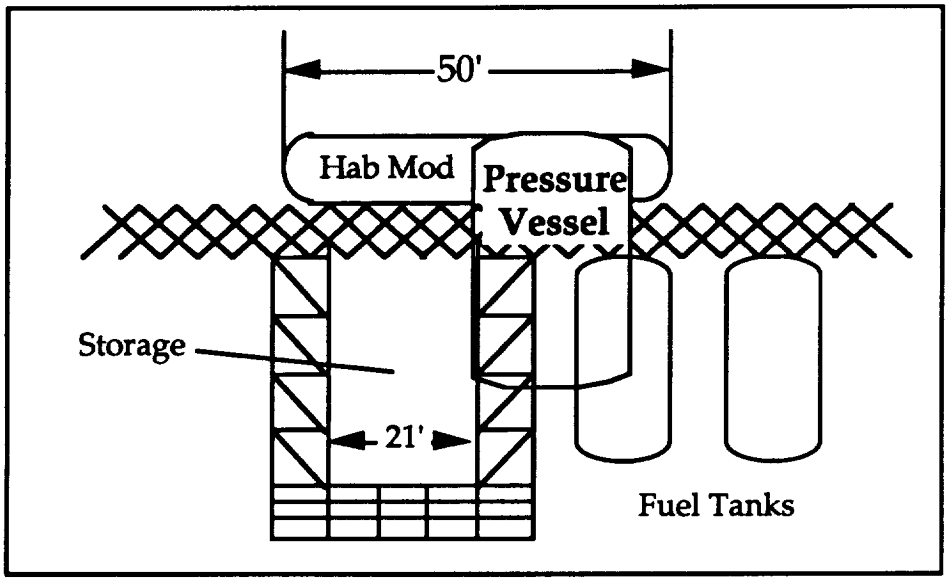

Figure 3.6.1 Storage Location

3.7 Robotic Track System

The robotic track system will be built to minimize EVA during station maintenance and vehicle assembly. There will be one 30 ft remote manipulator arm on each lunar bay and two on the Mars bay. There will also be two 60 ft manipulator arms on either side of the station to help with berthing and payload

transportation. The last manipulator arm will be located on the double fold deployable leading out to the solar arrays to help with storage, battery replacement, and solar array repair. That makes a total of seven manipulator arms in all. A typical track layout is shown in Figure 3.7.1

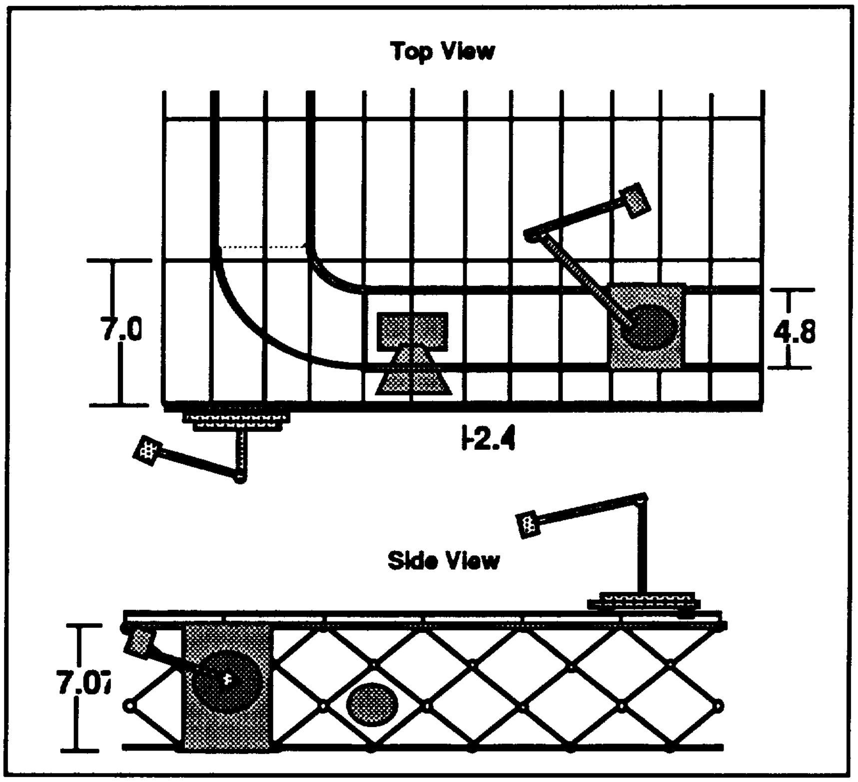

Figure 3.7.1 Track System

Each assembly bay will be encircled with a track to ensure robotic accessibility to all parts of the vehicle being assembled. The reboost and attitude adjustment jets are embedded within the truss so that vehicle and robotic movements will not be obstructed.

3.8 Docking and Berthing

The docking of the space shuttle will take place on the airlock without the pressurized sleeve. The docking mechanism will have to fulfill the following requirements

- withstand a force of 500 lb. created while docking with Shuttle.
- provide a rigid structural interface so station can correct attitude while the shuttle is docked.
- Provide an adequate amount of clearance for the shuttle.
- Be able to minimize docking loads.

The berthing of unmanned payloads will be done using a 60 ft robotic arm on the side of the station. Once the payload is within the 60 ft radius, the arm will berth with the payload and transport it to the appropriate bay.

3.9 Recommended Future Work

The future work for structures includes the following:

- docking and berthing conceptual designs.
- additional structural analysis of configuration using NASTRAN.
- complete construction scenario.
- astronaut mobility conceptual designs

4.0 Orbit and Attitude

4.1 Overview

In order to determine the altitude and inclination of SHARC's orbit, certain criteria were established. Most importantly, the AIAA competition requires that SHARC be able to accommodate a variety of launch vehicles. This limits the altitude and inclination to those that can be reached by the most vehicles. In addition, ISS has also mandated certain restrictions that extend beyond the AIAA requirements. We chose an inclination of 28.5° since this is the ideal inclination for a transportation node and is also easily serviced by the Space Shuttle, Delta, Atlas, and Titan. In addition, we discarded the option of having both stations fly formation since it would require an exact match of ballistic numbers or continual station keeping by SHARC.

Our altitude determination work was based mainly upon a TK Solver! program. The original program was modified several times during the course of our investigation. (The original program is found in Appendix A, section A.1.)

4.2 Altitude Determination

4.2.1 Use of Space Station Freedom for Support

ISS had originally thought to rely on Space Station Freedom (SSF) for crew, re-supply,Δ and medical support. To determine the altitude that would require the shortest time of flight (TOF) and the smallest change in velocity (ΔV) for a Hohmann transfer between the two stations, we utilized a TK Solver! program which calculated the total ΔV, the TOF, and the available launch windows. We only investigated the Hohmann transfer in order to minimize the

fuel required for transport flights. The results are shown in Figures 4.2.1, 4.2.2 and 4.2.3. (The program is listed in Appendix A)

Figure 4.2.1 shows that for any given altitude, the total ΔV required for a Hohmann transfer between SHARC and SSF is minimal. Since SSF is at an altitude of 400 km (217 n.mi.), this figure shows that as the altitude of SHARC increases to that of SSF, the required ΔV decreases. Therefore, from a ΔV standpoint, it is desirable to have the two stations as close in altitude as possible.

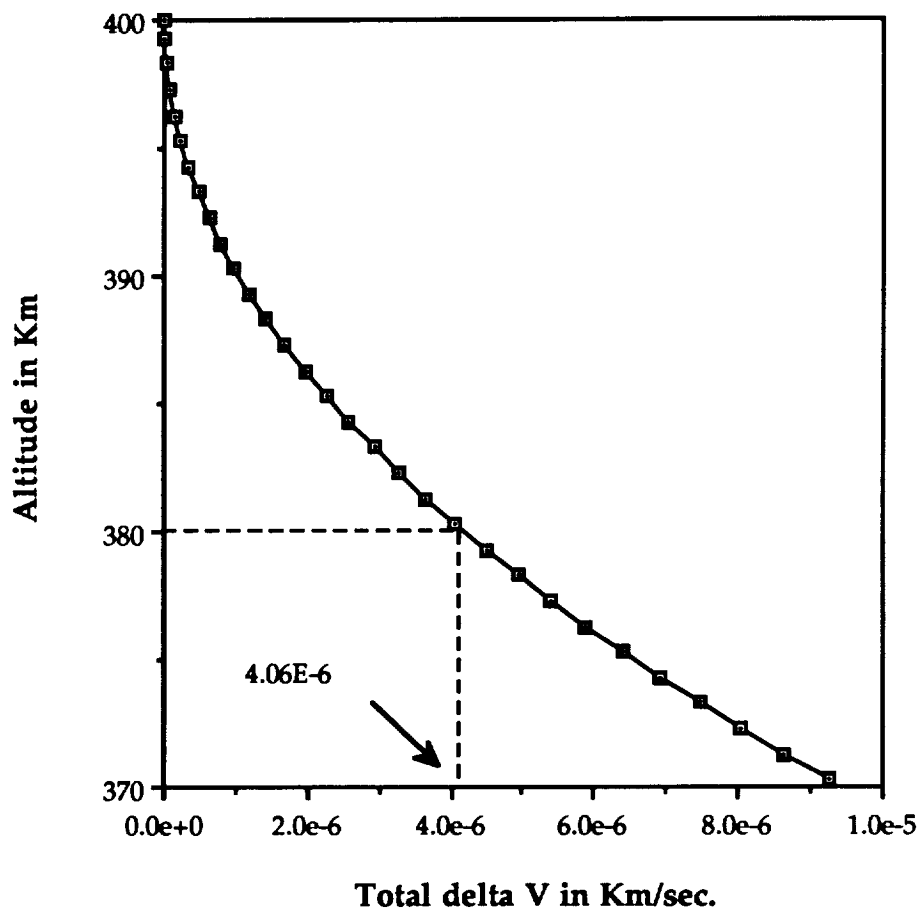

Figure 4.2.1: DV as a Function of Altitude for a Hohmann Transfer

Next we considered the TOF required for a Hohmann transfer. Figure 4.2.2 shows a plot of TOF as a function of altitude. As SHARC's altitude is increased to that of SSF, the TOF increases. However, since the TOF only varies from 46.13 to 46.28 minutes, it can be considered constant.

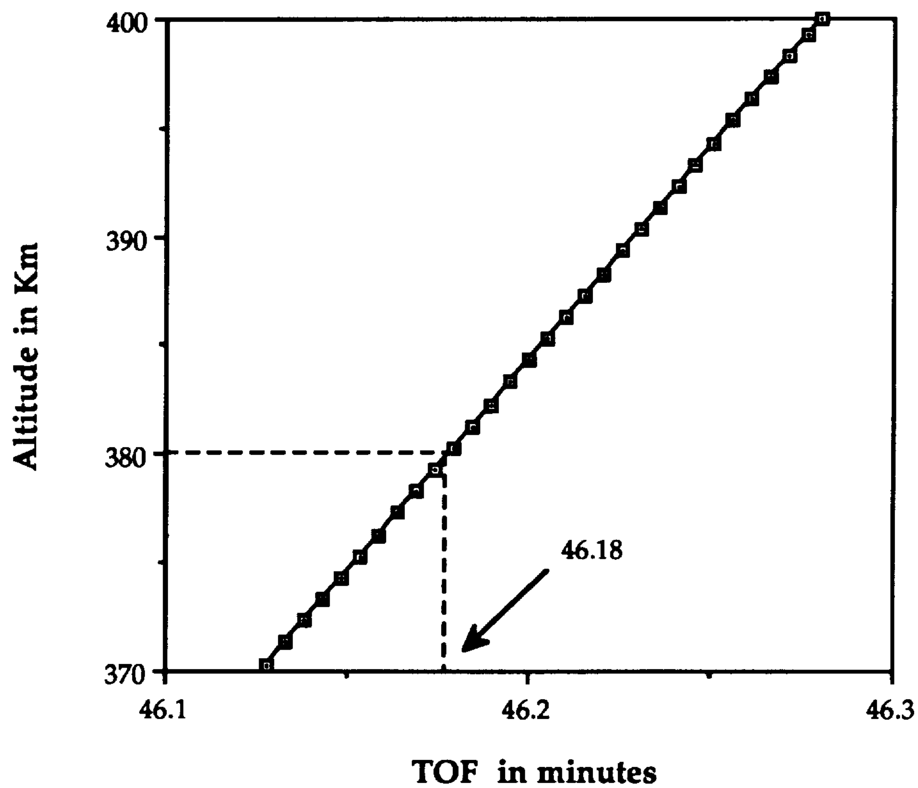

Figure 4.2.2: Time of Flight as a Function of Altitude for a Hohmann Transfer

Because the range of ΔV's is very small and the TOF is practically constant, we decided to investigate the availability of launch windows. Figure 4.2.3 shows the time between launch windows as a function of altitude.

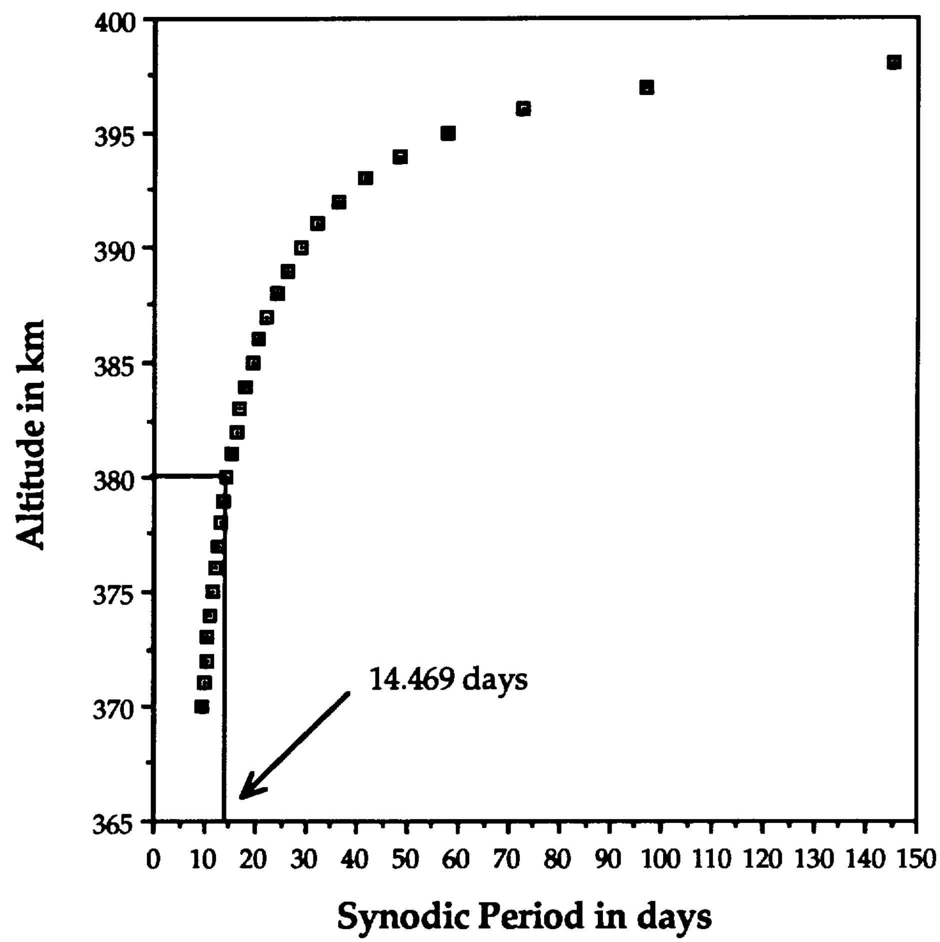

Figure 4.2.3: Time Between Launch Windows

From Figure 4.2.3 it is apparent that the minimum wait between launch windows is about 14.5 days. Worse yet, the closer SHARC comes to SSF, the larger the wait becomes. Therefore, it is highly unreasonable to rely on SSF since SHARC would be unable to wait 14.5 days for re-supply or medical support. In

addition, we also discounted the idea of a more direct transfer based on Lambert targeting since the transfer vehicle would require too much fuel. Also, should we have to evacuate some personnel, SSF would be unable to handle the extra crew members (See section 5.1.1) At this point in our investigation, we decided to abandon any reliance on SSF and to look instead at drag as the dominant factor in altitude selection.

4.2.2 Drag Forces

In order to determine the drag force acting on SHARC, we first needed to determine the coefficient of drag (C_d), the cross-sectional area, and the density at any given altitude. The C_d was determined to be 2.0 since air can be modeled as a rarefied gas at the altitudes we are investigating. In order to determine a worst-case scenario for drag, the cross-sectional area was taken to be that of the solar arrays (800 m^2), which was an the available estimate at the time of this analysis. Lastly, the density was calculated from a subroutine of program ASAP (See Appendix A for a listing). This data was then used as input in a TK Solver! program, which calculated a rough estimate for the drag force acting on SHARC as a function of altitude. This rough estimate was used in order to determine an altitude only. More refined calculations were used to determine the frequency for reboost (See Appendix E). Results are shown in Figure 4.2.4. (A listing of the TK Solver! program can be found in Appendix A)

Figure 4.2.4 shows the drag force acting on SHARC as a function of altitude. As altitude increases, the drag force decreases. Therefore, this plot shows that it is desirable to have as high an altitude as possible.

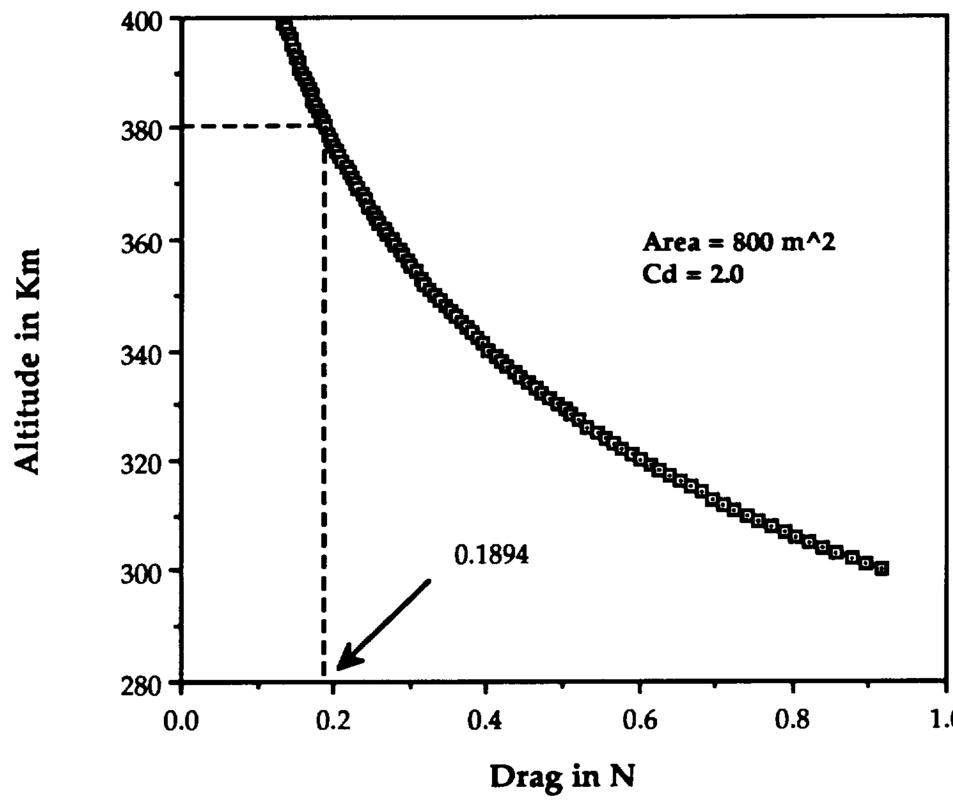

Figure 4.2.4: Drag as a Function of Altitude

Of more interest at this point, is the effect of drag on the semi-major axis (a) of the orbit. In order to determine the time rate of change of the semi-major axis (da/dt), the mass of SHARC was required. The mass was based on data for the "power tower" (see Section 3.2) but modified to include the extra habitation modules and the mass of the vehicles we are assembling. Since da/dt is a function of drag per unit mass, the TK Solver! drag program was modified to calculate da/dt as well as drag (See Appendix A for a listing). Since the drag force calculated was only a rough estimate, da/dt is similarly an estimate, used solely to aid in altitude selection.

Figure 4.2.5 shows da/dt as a function of altitude. As the altitude increases, the time rate of change of the semi-major axis decreases. This illustrates that the higher the altitude, the longer the time required between reboosts.

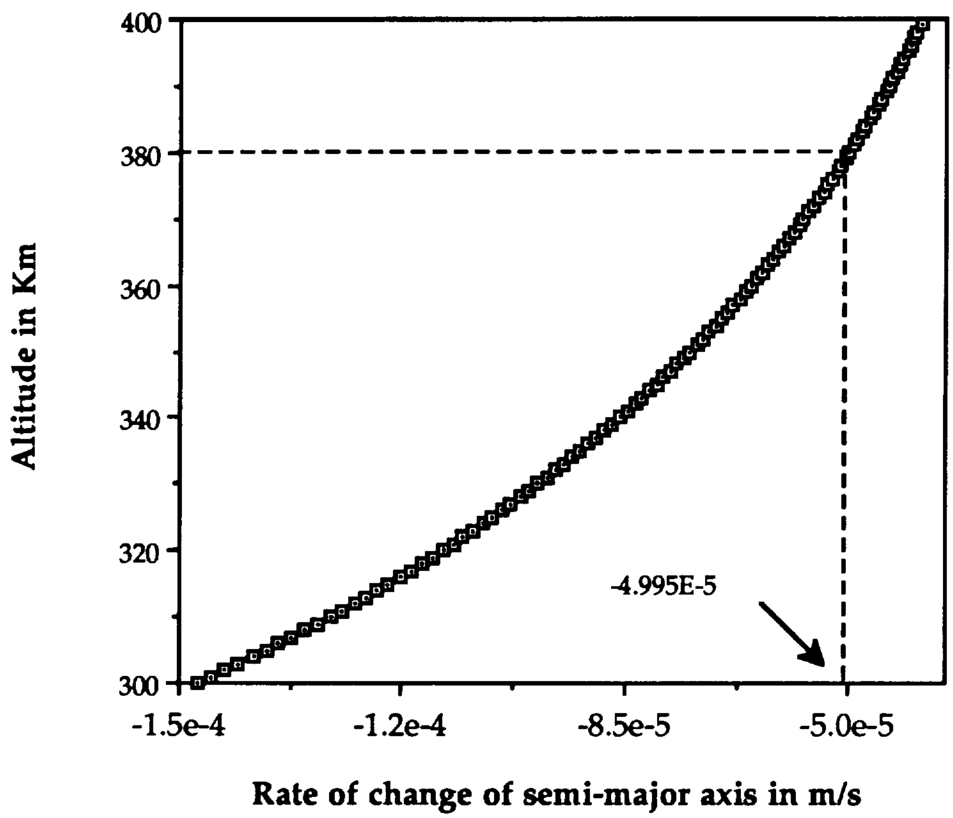

Figure 4.2.5: da/dt as a Function of Altitude

4.3 Orbital Elements

From this analysis, we have chosen an altitude of 380 km (205 n.mi.). This altitude in low enough for the Shuttle, Delta, Atlas and Titan to reach, but high enough to have low drag. The estimated orbital elements for SHARC at 380 km (205 n.mi.) can be found in Table 4.3.1. Drag and da/dt are estimated at 0.1894 N and 0.129 km per month respectively. Therefore, when SHARC decays in orbit, extensive station keeping should not be necessary. Our chosen altitude is indicated on all figures.

Table 4.3.1: Estimated Orbital Elements

Orbital Element	Value
Inclination	28.5°
Altitude	380 km
Orbital Velocity	7.680 m/sec.
Drag at 380 km	0.1894 N
da/dt	0.129 km/month
Synodic Period w/SSF	14.469 days
TOF to SSF	46.18 min.

5.0 Crew and Life Support

5.1 Crew Scenario

5.1.1 Original Assumptions

Obviously, the first thing that must be determined when designing a life support system is the crew scenario: how many people will be supported and for how long? Originally, SHARC was to be designed for a maximum twelve person crew with three months between resupply. This work crew would be supported on Freedom and travel our facility for 14 day work tours.

This scenario was to rely heavily on interaction with Space Station Freedom. After some orbital analysis, however, it was found that the launch windows between SHARC and Freedom will only open up every 10 days (see section 4.2.1). This made any sort of regular crew and supply transfer out of the question. Freedom's crew also consists of four members, and the added strain of more crew on their ECLSS is unreasonable. In addition, the small size of Freedom brought the whole concept of using it as a supply depot into question.

5.1.2 Revised Scenario

With all traffic to SHARC coming directly from Earth, the group began looking at resupply times and work tours. Since all crew would be lifted on the Space Shuttle, we had to estimate the number of launches per year coming to our station. NASA estimates that when the Shuttle reaches full operational capability, there will be a total of 15 launches per year. Recent experience, however, indicates that eight launches per year would be a more reasonable assumption. Of these eight, we decided that six would be bringing work crews to SHARC. We assumed that the Shuttle cargo bay would be used to transport

the life support supplies and any auxiliary payloads unrelated to our mission. This might include satellites for later deployment, or supplies for Freedom, or science platforms All assembly materials can be launched on an unmanned rocket and meet the crew at the station.

Since we are using the Shuttle for crew transport, a maximum of eight astronauts will be present on the station at any one time. We assumed that an average work tour would consist of five weeks: one week of powering up the station, three weeks of assembly operations, and one week of powering down. Each astronaut would work eight hours a day for six days a week. One hour per crewman per day would be maintenance of SHARC itself. The other seven would be assembly operations. This revised baseline was used to make the initial sizing estimates.

5.2 Sizing Estimates

With this 8-person, 35-day scenario, we began to calculate the equipment needed to support such a mission. The initial analysis was to a FORTRAN subroutine from the University of Texas (Dugan and Nottke, pp. 8-10). It sized a system for three cases:

- Open Loop System - no recycling, all supplies stored
- Partially Closed Loop System - wash water recycling only
- Closed Loop System - full air and water recycling

The data from this run is shown in Table 5.2.1. A FORTRAN listing is given in Appendix C.

Table 5.2.1 EC/LSS Size Estimates

	Open	Partially Closed	Closed
Power Required	1.560 kW	1.980 kW	3.580 kW
Waste Heat Generated	1.680 kW	2.580 kW	3.120 kW
Consumables Mass	9773 kg	1954 kg	507 kg
Hardware Mass	525 kg	612 kg	783 kg
Total Mass	10298 kg	2566 kg	1290 kg
Consumables Volume	10.083 m^3	2.671 m^3	1.313 m^3
Hardware Volume	59.332 m^3	12.876 m^3	5.868 m^3
Total Volume	69.415 m^3	15.547 m^3	7.181 m^3

Although it provided a valuable starting point, we felt that the program was not set up for a mission of this magnitude. Therefore, as a check, we used the more elaborate algorithm explained in Woodcock (pp. 201-205). This method was used to size the system for Freedom, so it would be suitable for SHARC.

The actual calculations used to get this estimate are shown in Appendix C. From this we determined the minimum resupply requirements for SHARC. Each trip, the astronauts would be required to bring up 146.9 kg of N_2 to replace air lost through module leakage and airlock use and 342.6 kg of food supplies. 107.2 kg of CH_4 would be produced as a byproduct of CO_2 recycling, and 182.5 kg of solid waste would have to be deorbited for disposal on Earth.

6.0 Power Subsystem

6.1 Introduction

The SHARC Electrical Power Subsystem (EPS) generates, stores, converts, regulates, and distributes electrical power. The EPS is an important factor in designing SHARC because nearly all other subsystems will require power to operate. In addition to the necessity of providing continuous average power for the duration of the mission, the EPS must exhibit:

- Capacity for periods of peak power
- Expandability for future power requirements
- Ease of servicing for quick repairs
- Utilization of existing technology
- Maximum durability for projected facility lifetime
- Good replacement characteristics for longer net lifetimes
- Safety considerations
- Minimum cost

Careful power system choices and integration with the overall facility design are crucial because the EPS has very concrete effects on SHARC's lifetime, attitude control, crew safety, and communications ability.

6.2 Subsystems Requiring Power

Besides the nominal subsystems that are described in this report, we have determined that SHARC will also require power for its unique mission operations. These include, but are not limited to:

- Exterior flood-lighting for spacecraft construction
- Specialized robotics and power tools
- Extra-Vehicular Activity (EVA)

6.3 EPS Design

The EPS was designed in five stages: identification of power requirements, selection of primary power sources, selection of energy storage systems, identification of power regulation, distribution, control systems, and integration. Integration with the overall design was a factor in all other stages and was considered in each decision.

6.3.1 Power Requirements

Identification of power requirements was necessary to define the amount of power the EPS must provide. We made rough estimates for the amount of power each subsystem required as well as estimates for the power required by the unique items mentioned previously. We estimated peak power needs, minimum power needs, and average needs along with a 10% safety allowance for electrical inefficiency losses (due to frictional heating, deterioration, etc.). Table 6.3.1 presents the peak and average power requirement figures

Table 6.3.1 Subsystem Power Requirements

Subsystem	Peak Power (kW)	Average Power (kW)
ECLSS	20.4	17.0
Thermal	0.0	0.0
Propulsion	.01	.01
EPS losses	10% of total	same
Robotics	17.5	7.5
Power Tools	4.0	2.0
Communications	5.4	4.0
Lighting	7.7	4.0
Sensors	.2	.2
Airlocks	.4	.2
EVA	6.2	3.0
Total	68	41.7

6.3.2 Primary Power System Options

Because one of our assumptions was the predominant use of existing or near term technology, primary power source selection required consideration of available technology and what each option had to offer. Since SHARC has a large power requirement, we considered only systems capable of handling this demand. Other factors we considered were:

- Power capacity
- Material and installation cost
- Lifetime and durability
- Stability and maneuverability
- Low orbit drag
- Sensitivity to Sun angle and shadowing
- Fuel re-supply, if necessary

Five possible candidates for large scale power generation were photo voltaic, solar-dynamic, large-scale nuclear systems, tethers, and microwave power beaming. Tethers and microwave beaming were immediately eliminated because of their experimental status. We also eliminated nuclear power systems because although there is active research being done on space-qualified nuclear reactors (such as the SP-100 project), the concept is still in the experimental stage. There may also be crew safety problems and possible environmental consequences with nuclear power in a low earth orbit. Solar-dynamic systems had impressive performance characteristics but were eliminated because of the immaturity of the technology. As with Freedom, solar dynamic systems will probably be used as an evolutionary technology to be implemented as it becomes available.

The system that met most of our requirements was photo voltaic solar arrays. Photo voltaic systems are a well-proven and reliable technology with a considerable mission database from which we can predict lifetimes and performance characteristics. They are relatively easy to deploy and, with careful construction, do not require thermal conditioning Their largest disadvantage is the low efficiencies of the silicon cells (a current estimate for silicon solar cell efficiency is 11-12%) which leads to large array sizes for a given power requirement.

Silicon is the current material of choice for cell construction. Although there are several advanced materials which exhibit improved durability and efficiencies (e.g. Gallium-Arsenide and Indium-Phosphide), these types of arrays are still being tested and have not been space-qualified. They would also be considerably more expensive than silicon arrays.

The calculations to size a silicon solar array system are in Appendix D. The results show that to provide 62kW of power, SHARC requires a set of arrays 1854 m2 in total area and 2267 kg in mass. The configuration of these arrays is shown in Figure 1.3.1. There are 8 pairs of folding array panels deployed along an erectable truss. Each panel has approximate dimensions of 13'x95'. The projected lifetime of the arrays (their service time before they need to be replaced) is roughly 10 years, after which time the arrays will experience a degradation of roughly 25% in conversion efficiency.

6.3.3 Secondary/Storage Power Options

In choosing a power storage system, the considerations were:

- Supplementing primary power during peak loading
- Providing power for vital components during possible power failures
- Providing power during solar eclipsing or shadowing

Batteries and rechargeable fuel cells were considered as possible power storage sources. Because SHARC has a long mission lifetime of 30 years or more, fuel cells were eliminated as the main secondary power system because of fuel re-supply problems. Regenerative fuel cells were eliminated because of their experimental status. Therefore, rechargeable batteries were chosen as the main secondary power supply.

The next step was to choose from the wide variety of batteries available. There are several battery characteristics which determine their performance. For our purposes, the most important were:

- Depth of discharge (DOD) curve
- Cycle life
- Amp-hour or watt-hour capacity
- Energy density

With all other things being equal, a battery which exhibits the highest values for these parameters would show the best performance and have the longest cycle life. Good performance and durability are important because the environment that the batteries must operate in is considered extremely harsh. For example, LEO spacecraft encounter at most one eclipse period each orbit or about 15 eclipse periods per day, with maximum shadowing of approximately 36 minutes. Therefore, the batteries must charge and discharge about 5,000 times each year. (Wertz, p. 362).

We set a requirement that the cycle life of the battery be at least 5000 cycles (approx. one year), because replacing batteries with short lifetimes would quickly become laborious and expensive. This requirement alone eliminated many types of batteries. Three battery types that do have potential cycle lives of 5000 cycles or greater are: Nickel-Cadmium (Ni-Cad), Nickel-Hydrogen (Ni-H_2), and Silver-Hydrogen (Ag-H_2) batteries. Ag-H_2 batteries were eliminated because they have not been space qualified. Note that Ni-H_2 batteries have been space qualified for geostationary orbit (GEO), but not for

LEO. Since SHARC will not begin construction until 1998, we assumed that this technology will be available for LEO by then. Thus, we did not eliminate Ni-H_2 batteries because of their newer status.

Table 6.3.1 presents some of the parameters described earlier for Ni-Cad and Ni-H_2 batteries (Wertz,p. 362).

Table 6.3.1 Performance Characteristics for Ni-H_2 and Ni-Cad Batteries

Battery system	Cycle life (dep. on DOD)	Energy capacity (amp-hours)	Energy density (Watt-hours/kg)
Nickel-Hydrogen (Ni-H_2) (individual pressure vessel)	400-40,000	insufficient data	25-40
Nickel-Cadmium (Ni-Cad)	300-25,000	5-100	25-30

From this data, we determined that Ni-H_2 batteries exhibit the best combination of lifetime and performance. Appendix D contains the calculations for the number of Ni-H_2 batteries required to supply 62 kW of power during the eclipse period of orbit. These batteries will have a capacity of 100 amp-hours and an energy density of 25 W-hrs/kg. The battery containers will be individual pressure vessels with the Nickel and Hydrogen electrodes configured inside the vessels in a stacked disk design. The vessels will then be placed in large, thermally controlled cases between the solar arrays (see Figure 1.3.1).

Assuming a worst case DOD of 50% (for peak power), SHARC will require 27 Ni-H_2 batteries connected in parallel for maximum capacity and redundancy. They will also be charged in parallel for simplicity and minimum cost. The mass of each battery will be approximately 112 kg. For the worst case DOD, the lifetime of these batteries will be approximately 2 years. However,

SHARC will usually require the lower average power of 42 kW. In this case, the DOD for these batteries can be reduced to nearly 30%. This is important because reducing depth of discharge increases cycle life. At a DOD of 30%, we can expect a lifetime of nearly 5-6 years (Wertz, p. 363)

6.3.4 Power Regulation, Control, and Distribution

Power regulation and control refers to controlling the solar array, regulating the bus voltage, and charging the battery. Power distribution includes the cabling, fault protection, and switching gear to turn the power on and off to the spacecraft loads. A baseline schematic of the EPS regulation, control, and distribution is shown in Figure 6.3.1 (Rauschenbach, p. 10).

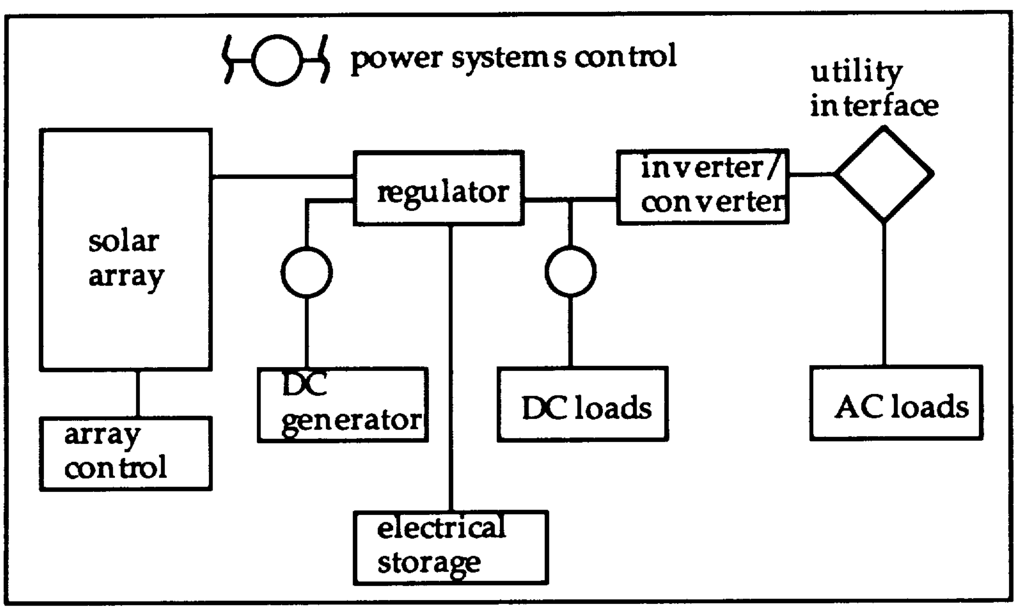

Figure 6.3.1 Power Control, Regulation, Distribution Schematic

The solar array and the array control units encompass all the elements necessary to support the array such as the mounting and the motors required to control the movement of the array as it tracks the sun. The regulator distributes the power from the array to the loads and battery, controls the operation of the

array (turning it on and off), and maintains the proper array and bus voltage. It also provides fault detection by shutting the system down in the event of abnormal conditions such as large voltage spikes. The inverter/converter converts the variable voltage DC to constant voltage AC for those loads requiring it.

The nominal voltage of the distribution system will be a standard 28 Volt bus because of its well-proven status, reliability, and safety. The solar arrays will operate at a nominal 33 Volts to create a potential for charging the batteries. The distribution system will also be centralized. This means that the converters will be placed out at each load end separately. The advantage of this system is that the EPS does not have to be designed for many different applications and can handle loads with many different voltages, as SHARC is expected to have.

6.4 Recommendations

The calculations performed in this section are based upon data obtained from many sources. As is usually the case when using many references, much of the information was vague and contradictory. An effort has been made, when using conflicting values, to use data that has been confirmed by two or more sources. In addition, the most recent values available were used whenever possible.

For future work, there should be more concentration on methods for erecting the solar arrays and to determine, through structural analysis (e.g. NASTRAN), whether the solar arrays can withstand the forces due to attitude control and reboost (our current plan is to simply retract the arrays during reboost). For the battery system, more research needs to be done on Ni-H_2 battery design, performance, and arrangement. Finally, because of the time constraints, not much could be done with respect to the power regulation,

distribution, control systems. Future work can concentrate on refining the chosen systems and their placement in the overall structure.

7.0 Robotics

7.1 Overview

Orbital assembly operations will require large amounts of telerobotic hardware. Without it, SHARC crewmembers would have to perform large amounts of EVA, creating unnecessary risks. We have identified four general tasks for the station robots:

- Assembly of space structures
- Space station maintenance and repair
- Satellite and spacecraft servicing, repair, and assembly
- Maintenance of other robots

7.2 Applications of Robotics Systems on SHARC

7.2.1 Assembly of space structures

The first assembly task we must consider is SHARC itself. This involves a wide spectrum of tasks, from the assembly of large modules to small mating tasks such as bolting and locking. This assembly process could be highly structured to minimize the level of uncertainty.

7.2.2 SHARC Maintenance and Repair

SHARC will require continuous inspection for fatigue failures, flaws, meteorite damage, etc. The structure will be made up of very lightweight and specialized materials experiencing high radiation levels, thermal shocks, and cyclic vibrations. Shuttle missions average 4 to 6 failures per day of operation. Repairing these failures is a major drain on crew time. On SHARC, however, robots should be able to perform these tasks.

Initially, SHARC will use telerobotics to inspect the hull and structure, with any repairs done by EVA. Although not possible with current technology, it should eventually become possible for the robots to perform repairs without human guidance.

7.2.3 Satellite and spacecraft servicing and assembly

The primary function of SHARC is the orbital assembly and servicing of spacecraft, satellites, payloads, and other station elements. These operations will include the following:

- Maintenance and repair
- Berthing and Docking
- Resupply
- Refueling
- Assembly

Servicing includes all activities associated with restoring the operational capability of a system including fault identification and diagnosis, planned maintenance, and corrective maintenance. Specific servicing tasks include inspection, fault isolation, refurbishment, replacement of parts, checkout, calibration, and repair. In addition, the current mission scenario is for unmanned launch vehicles to dock with the station on a regular basis. This will require sophisticated robotics systems to allow the incoming payload to be moved into place without damaging the station's trusswork. Finally, SHARC's assembly bays will be equipped with robotic arms of various sizes and sophistication to allow vehicle assembly to take place without excessive EVA.

7.2.4 Maintenance of robots

Robots in space will generally be lightweight and intricate. Therefore, the robots designed by Tesar follow a modular approach. Damaged or unwanted systems could simply be removed and replaced by another module. This modularity should increase system versatility and reliability. The desired downtime for the robots will be about 2% of total operating time. Two thirds of this downtime will be for regular maintenance and one third for emergency repairs.

Generally, robotic systems will degenerate with use and system parameters will change. Some of these changes can be dealt with directly by self-diagnosis and corrections to the operating software. In the case of structural damage, a second service robot will repair the damaged robots using new modules. This type of robot is still under development, however, and may not be available for initial operations.

7.3 Requirements for Robotic Systems

Basic requirements for space telerobotic systems were identified in the 1985 report by Tesar:

7.3.1 Multi-task capability

The more distinct tasks a given robot can perform, the fewer the robots that will be necessary to operate the station. The variety of tasks the robot systems need to perform suggests generic multi-purpose robots with an ever-increasing level of flexibility.

7.3.2 Level of machine intelligence

The full array of chores, inspection, maintenance, and response to emergencies will overload the personnel on board the station. A high level of machine intelligence for the robotics systems will help to alleviate this problem.

7.3.3 Time efficient operation

The time efficient operation of the supporting robotics system is an important criterion for its design and implementation. The need for time efficient operation is highlighted by the fact that the shuttle has 4 to 6 failures per day, and docking with a satellite now requires 8 to 10 hours.

7.3.4 Unstructured task level

Many uncertainties will exist because of the differences of "as is" versus "as designed", resulting from imperfect assembly, maintenance, parts replacement and updates, structural damage, etc. The goal is to reduce the level of numerical uncertainty to a minimum.

7.3.5 Geometric dexterity

The minimum dexterity required to control spatial motion is 6 DOF; however, extra DOF (say a total of 8) make a wider range of motions feasible. It is conceivable to add extra DOF modules to a robot to enhance its dexterity on demand.

7.3.6 Portability and Mobility

A major issue for SHARC is to establish the ability to move about the station to perform planned or emergency repairs and to perform assembly and disassembly tasks. There are three approaches for mobility in space operations:

- Rail transport
- Crawling
- Free flight

In the full operational phase of the station, a combination of all three of these concepts will probably be employed.

7.3.7 Precision and load capacity

Many operations in the station will require high levels of precision (1 to 10 thousandth of an inch), even when the robot structure is disturbed by forces generated by the process being performed. The precision requirement for a robot that is under a load will increase the robot's weight. Lightweight robots which can maintain precision under load need to be developed.

7.3.8 Reliability

Robots for SHARC will have to operate in vacuum, in radiation, experience thermal gradients, and be impacted by micro-meteorites. Nonetheless, these robot systems must be as reliable as possible. Redundancy in some of the hardware components and robots made of modules which could be replaced easily can make the robotic system more reliable. Unfortunately, the need to be lightweight and compact makes reliability more difficult to achieve.

7.3.9 Obstacle avoidance

Since the operating environment in the assembly and service area will likely be constrained and cluttered with obstacles, collision avoidance technology must be part of the operating software of these robot systems.

7.3.10 Force sensing

The force level experienced at the end-effector of a robot is critical to determine whether a given task is being performed properly, to determine if damage is occurring to the part being manipulated, or to be aware of excessive forces in the robot itself. Force feedback to the human operator is necessary to assist him in carrying out complex operations.

7.3.11 Smoothness of operation

Smooth operation of robot systems means that a minimal amount of dynamic shock occurs either in the command signals of the robot, at its end-effector, or within the structure of the robot itself. Dynamic shock leads to vibrations which would impair the operating precision of the robots.

7.3.12 Operational envelope

The present Remote Manipulator System (RMS) of the Shuttle has a 55 ft. reach and a level of dexterity similar to the human arm. This serial structure is ideal for low precision deployment functions. Beyond these, smaller scale systems should be developed. The scales for the operational envelope might be:

RMS	60 ft.
MRMS	30 ft.
Man sized	5 ft.

7.3.13 Vision

Vision has the same importance to SHARC robotics systems as a feedback mechanism. Its principal function will be to enable continuous and autonomous inspection of the space station by using the data base for reference. Vision is also the dominant means of feedback for the operator to rapidly access the global condition of a work scene. Today it is possible to use fiber optics in the finger tips of end-effectors to make very close inspection feasible.

7.4 Robotics systems selected for SHARC

The robotics system designed for SHARC will be similar to the one designed for Space Station Freedom in many aspects. The Space Station Remote Manipulator System and Flight Telerobotic Servicer are suitable for SHARC's mission in that they are designed to minimize EVA and to help assemble components and structures. By utilizing Freedom's designs, the cost to build SHARC's system can be reduced. The robotics systems selected will be mainly telerobotic. Telerobotic servicing has been accomplished in earth applications, such as nuclear power plants. Advantages of telerobotics include:

- Availability of human decision making, adaptive reasoning and problem solving without the hazards associated with placing a human at the worksite.
- Reduced demands for human operator time as compared with EVA.
- Ability to perform in conduction with EVA.
- The capability for fully repetitive actions.

However, telerobotic servicing systems have the following disadvantages:

- Limited capability (dexterity, reach, controllability) associated with existing telerobotic technology.
- Demands on the human operator.

The state-of-the-art in telerobotic systems and equipment for space vehicle servicing was described in the NASA JSC Servicing Equipment Catalog (JSC-22976, 1988). The systems which will be integrated into SHARC are the following:

7.4.1 Light-Weight Module Service Tool (LW/MST)

LW/MST is a device to permit remote on-orbit exchange of On-orbit Replaceable Units (ORUs) when coupled to an automated servicer system. It can be redesigned for use with Remote Manipulator System and other manipulator systems. This tool will permit on-orbit exchange of spacecraft module, payloads, and instrument orbital replacement units.

7.4.2 Payload Berthing System (PBS)

PBS provides on-orbit docking/berthing of payloads for servicing, repair or temporary holding. The PBS is sidewall mounted at the primary attachment locations of the cargo bay.

7.4.3 Servo-Actuated Manipulator System with Intelligence Networks (SAMSIN)

SAMSIN is a bilateral force reflecting master-slave servo manipulator. A general purpose electrical-mechanical device, SAMSIN is used to extend the hand and arm manipulative capacity into a remote and hostile environment.

7.4.4 Standard End Effector (SEE)

SEE is the terminal device on RMS arm or Flight Telerobotic Servicer, and its primary function is to capture, hold, and release payloads. For SHARC, it is desired that special purpose end effectors (welding, drilling, claming, etc.) be developed to meet the servicing requirements.

7.4.5 Universal Servicing Tool (UST)

UST is a flight power tool that allows changeout of the tool attachments on orbit. Designed to anchor itself to a payload or spacecraft module, the UST can be used to remove or tighten bolts, and operate latches and fasteners while reacting the resulting torque to the anchor point.

7.4.6 Remote Manipulator System (RMS)

RMS is a mechanical arm which augments the Shuttle systems in performing the deployment and/or retrieval of a payload. In addition, the RMS may be used for other tasks in extravehicular activities or cargo transfer on SHARC. This system will be described further later in the report.

7.4.7 Flight Telerobotic Servicer (FTS)

The FTS is designed to be a teleoperated device controlled by a crew member from within SHARC. Limited autonomous capability is projected. The two principle components are the telerobot and workstations. It will be discussed further later in the report.

7.4.8 SHARC Remote Manipulator System

The SHARC Remote Manipulator System will be similar to the one designed for SSF. The Freedom RMS is designed by the Canadian Space Agency. The main differences between SHARC's and Freedom's RMS will be that the actuators, computers, and joints on the SHARC RMS will be more advanced; however, the specific component design is beyond the scope of this report. The technical specification cited will be that of SSF RMS with some minor adjustment. Nevertheless, the basic design concept is similar: lightweight, high payload, modular, and precise.

The uses of RMS can be categorized as the following:

- SHARC construction, assembly, and maintenance
- Payload handling and servicing
- Capture and handling of free flyers
- Support for extravehicular activities

The two kinds of RMS on SHARC will be 18.3 m (60 ft.) and 9.1m (30 ft.) long telerobotic arms to be used for handling large objects on the Space Station. It consists of seven joints, two latching end effectors (LEE), two boom assemblies, two arm computer units (ACU), video cameras, and associated equipment. The RMS configuration for SHARC will be similar to the one in Figure 7.4.1.

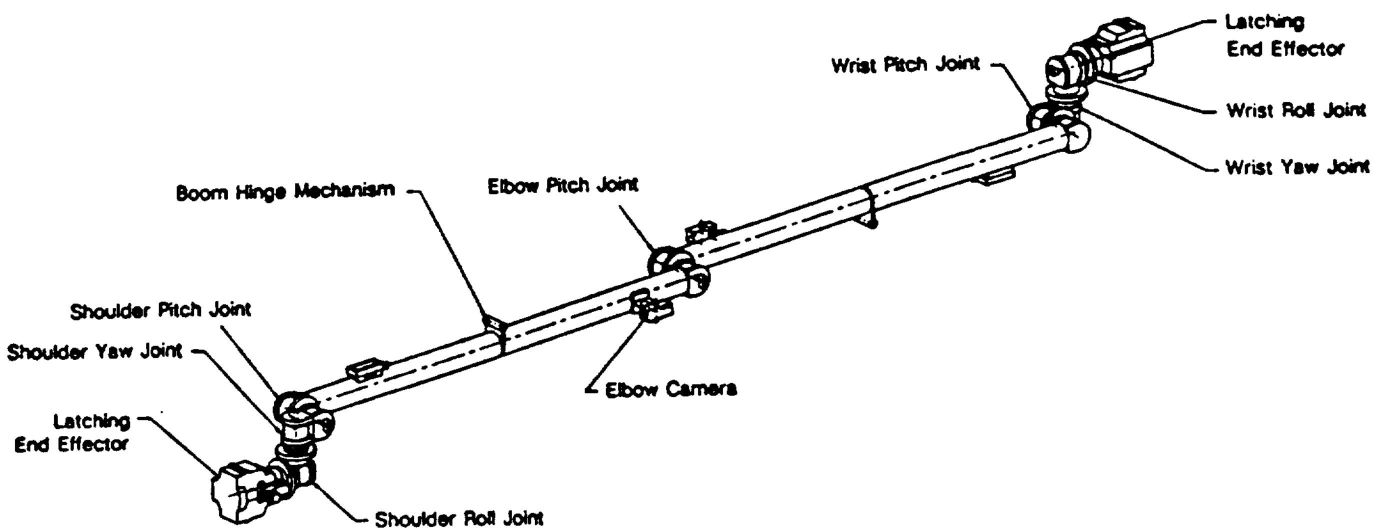

Figure 7.4.1 Physical Configuration of Remote Manipulator System

The seven joints, each representing a rotational degree of freedom, provide maneuvering and positioning capabilities. The joint will be modular, helping to reduce the number of spare parts. In addition, any future changes or improvements of the joints can easily be connected to existing joints or the LEE. The LEE at the base provides structural and electrical (power and data) interfaces to SHARC. The tip LEE is used for payload capture and release.

The physical characteristics of the RMS are the following:

- The RMS is to operate in the extravehicular environment of SHARC. The Mobile Remote Servicer Base System (see section 3.7) will be used as the base for SHARC RMS.
- The tip end effector is compatible with the SRMS-type Grapple Fixtures defined in NSTS 07700.

- The capture operation of the RMS will accommodate the following misalignment of the grapple probe: (based on SSF RMS)

Linear misalignment	= 0 to 0.1 m axial direction,
	$\pm$ 0.1 m radial direction
Angular misalignment	= $\pm$ 10 degree roll,
	$\pm$ 15 degree pitch and yaw

- The specified performance of the RMS is listed in Table 7.4.1.

Table 7.4.1: SHARC RMS Performance Requirements (Kumar & Hayes)

Payload Size			Velocity		Stopping Distance	
Mass	Length	Diameter	Linear	Rotational	Linear	Rotational
0 kg	-	-	0.37 m/s	4 deg/s	0.61 m	3 deg
20,900 kg	4.5 m	17.0 m	0.022 m/s	0.24 deg/s	0.61 m	3.8 deg
116,000 kg	24.1 m	34.3 m	0.012 m/s	0.04 deg/s	1.09 m	5.7 deg

- The power requirements usage for RMS is 1800 watts average and 2500 watts peak. The data transfer requires two 1553B data buses. The video capability will be stereo vision to help the operator understanding the work environment.
- The RMS is a single failure tolerant design, with automatic safing following any failure.
- The SHARC RMS is designed to operate on orbit for 30 years with periodic maintenance and refurbishment.

7.4.9 SHARC Flight Telerobotic Servicer (FTS)

The SHARC Flight Telerobotic Servicer is similar to the Flight Telerobotic Servicer developed by Martin Marietta for Space Station Freedom in many aspects. The main difference will be that the SHARC FTS will be at least 5 years more advanced than the Freedom FTS.

The SHARC FTS will have the basic capabilities to support any task it might be assigned, although the design is derived from seven specific design reference tasks:

- Install and remove truss members
- Install a structural interface adapter on the truss
- Change and replace orbital replacement units
- Mate thermal utility connectors
- Perform inspection tasks
- Assemble and maintain the electrical power system
- Light and precise assembly operations of major spacecraft

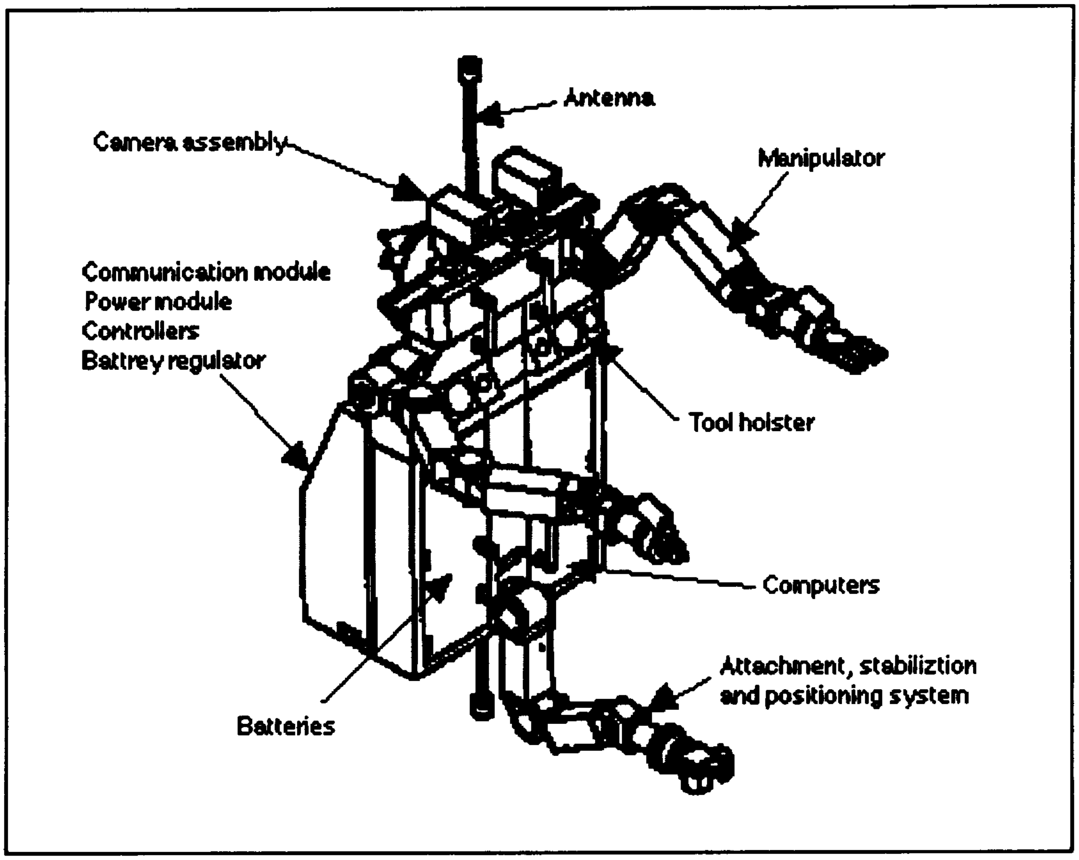

Figure 7.4.2 SHARC Flight Telerobotic Servicer

The SHARC FTS (Figure 7.4.2) has two 5′ manipulators, each with seven degrees of freedom (DOF). It also has one 5-DOF attachment stabilization and positioning system mounted on a compact body, which serves as a leg support. The body contains internal electronics which provides the power, data management, processing, and communication functions. The internal components, manipulators, and leg are modular orbital replacement units. A camera positioning assembly with two stereo-vision cameras, two antennas, and storage locations for tools and end effectors are mounted on the body as well.

The FTS has three operating modes: dependent, transporter-attached, and independent. When dependent, the telerobot is attached to an worksite attachment fixture or through an umbilical to receive and transfer power and

data. In the transporter-attached mode, the telerobot can be operated from SHARC manipulator, while being connected directly to utilities through the RMS. In the independent mode, the FTS will derive power from internal batteries and data and video signals through its antennas. It only needs mechanical attachment to the worksite, which gives it the flexibility to perform tasks at worksites without utilities.

There will be workstations on SHARC dedicated to the FTS. On SHARC, the workstation is the man-machine interface to the FTS, providing the displays and controls that permit the FTS to be operated by an individual. The FTS can be teleoperated through master hand controllers with slave manipulators. The FTS operator will have the option of using voice commands to operate the FTS to perform simple tasks like inspection of trusses or in conjunction with teleoperation.

Some technical specifications for the FTS are the following:

- The total weight of the telerobot and the workstation will not be more than 1500 lb. The power consumption will be 2000 W peak power, 1000 W average power, and 350 W standby power.
- The FTS will have a system accuracy of less than 1.0 in. in position and $\pm$ 3.0 degree in orientation.
- The two manipulators (arms) have a repeatability of less than 0.005 in. in position and $\pm$ 0.05 degree in orientation. The incremental motion of the manipulators is less than 0.001 in. and less than 0.01 degree at the center of the tool plate.
- All FTS processors access 1553b networks and are based on 80486 technology.

- The communication system consists of a Ku-band or optical receiver for video/ telemetry/command data transceiver and the EVA safety shutdown functions of a transmitter and EVA receiver.
- The operator will have the capability of selecting and defining coordinate frames, and he/she will be able to perform dual-arm coordinated control of a grasped object with a single hand controller. The control algorithms provide a smooth, safe transfer between autonomous and teleoperation control.
- The FTS is designed for growth and evolution over the years. A functional architecture NASA/NBS Standard Reference Model for Telerobot Control System Architecture will be supported by the software and computer architectures so that orderly expansion can be accomplished.

7.5 Future work

Future research will be focused on the research and development of SHARC RMS and FTS subsystems compared with those of Freedom. Since the systems designed for Freedom are more than five years old, integration with most advanced technology to upgrade the RMS and FTS for SHARC is highly desirable.

8.0 GN&C/Reboost

8.1 Propulsion Requirements

SHARC will require some form of propulsion for station keeping and attitude adjustments. With regard to reboost, the calculated rate of descent for the current SHARC configuration is eight kilometers every month (see Table 4.3.1). This gives an indication of how much fuel is needed over 30 years. In addition, the propulsion subsystem must be able to respond to these factors:

- Reboost
- Attitude control
- Avoidance of large orbital debris

8.2 Design Considerations

In determining the propulsion requirement of SHARC, several design considerations were addressed. These considerations were:

- Reboost time
- Refuel period
- Propulsion equipment (propellant tanks, lines, thrusters or engines, and pressure-regulation)
- Thruster location (structural limitations)
- Mass
- Power requirements
- Cost

8.3 Types of Propulsion Systems

Propulsion systems are divided into two categories: chemical and non-chemical. Chemical propulsion systems are:

- Solid chemical propulsion systems
- Liquid chemical propulsion systems
- Gaseous chemical propulsion systems
- Hybrid propulsion systems

Non-chemical propulsion systems are:

- Fluidic Momentum Controller
- Large Area Magnetic Torquer
- Ion engines and magnetoplasmadynamic thrusters

8.4 Propulsion Subsystem Design Process

8.4.1 Assumptions

In choosing the attitude control and reboost system, we assumed that the Phobos Transfer Vehicle and two lunar transfer vehicles would be present. This was a worst case scenario in which the mass of SHARC would be at a maximum, and most of the calculations used in determining propulsion values (see Appendix E) were based on this assumption. The equations are derived from Wertz (Chapters 6.3,11,17).

8.4.2 The Propulsion Design Process

Wertz suggests a nine-step design process in determining the most effective propulsion system for SHARC. These steps are:

1. Determine the primary function of the propulsion system.
2. Calculate the ΔV's the system must deliver.
3. Estimate the maximum thrust the structure can withstand.
4. Select the type of engine (solid, liquid, etc.) that best suits 1 & 2.
5. Choose a specific impulse (I_{sp}) within the range for the chosen type of engine.
6. Use the thrust and mass data to estimate the engine mass.
7. Use analytical equations to estimate propellant mass.
8. Calculate the total impulse using the I_{sp} and propellant mass.
9. Choose a system that satisfies all previous criteria.

The steps we considered to be the most important, 1, 2, and 7, will now be described.

8.4.3 Step One - Determine Primary Function

In determining the propulsion system, the first step was to specify the primary requirements of the SHARC system. These functions are:

- Reboost after period of sixty days
- 3-axis attitude control during normal orbit
- 3-axis attitude control during reboost
- Rotate ninety degrees for reboost

8.4.4 Step Two - Calculate ΔV Requirements

After the primary functions of the station were specified, the ΔV required for each function was determined. For the reboost time, a period of sixty days was considered. Sixty days was chosen as the time necessary for refueling by the STS Orbiter. The program ASAP (see Appendix E) was used to determine the altitude, 364 km, after the reboost time. The ΔVs for a Hohmann transfer were calculated from 364 km to 380 km. The total change in velocity required was approximately 9.1 m/s.

For attitude control about 3-axes during normal orbit, we anticipated a torque due to drag of 5 to 10 N-m. This torque arises due to the difference in location between the center of the effective area and the center of mass. When the orientation of SHARC is local vertical, torque due to the gravity gradient is assumed to be small compared to drag. During reboost, when SHARC is rotated ninety degrees, a torque due to the gravity gradient effect was the primary external disturbance torque considered. Nearly 50 N-m of torque was calculated for the worst case scenario. Using these values a system with 100K pulses at 0.2 seconds per pulse was considered.

8.4.5 Step Seven - Determine Propellant Requirements and Mass

A propellant budget was calculated using the ΔV budget. The propellant requirement for one reboost was 4870 kg. An estimate for the attitude control propellant requirement was 1100 kg, approximately 22% of the reboost fuel. The 22% estimate takes into account attitude control during both reboost and normal orbit revolution. This resulted in a total propellant requirement of approximately 5970 kg for a period of sixty days. Finally, a safety factor of two was taken into account which resulted in a total propellant requirement of 11940 kg.

8.5 Propulsion System Chosen

For SHARC's propulsion system, only liquid chemical systems were considered. The Fluidic Momentum Controller (FMC) was not selected because of difficulties in integrating it with the truss structure. The Large Area Magnetic Torquer was not selected because it was limited to a certain orientation of the dipole moment with respect to the Earth's magnetic field. A continuous torque would not be available. As for the ion engines and the MPDs, insufficient information was available on off-the-shelf models. Also, because ion engines are a new technology, their reliability is questionable.

Solid chemical propellant could not be used, because complete control of the thrust output was not possible. Gas propellant systems were not considered because of their heavy mass and low specific impulse. Hybrid propellant systems do have some advantages, but their reliability depends on how the system was designed. The feasibility of a large thrust hybrid system has not been determined. Ultimately, a liquid propellant system was chosen because of its controllability and good performance.

8.5.1 Propellant and thrusters

After following this design process, a propulsion system for the SHARC station was finally determined. A hydrazine (monopropellant NH_4) attitude control system was chosen for its simplicity and good NH_4 decomposition characteristics. The attitude thrusters are based on the GRO spacecraft propulsion system. Each thruster provides a maximum of 30N and a propellant specific impulse of approximately 220 seconds. A N_2O_4/MMH system was chosen for reboost. One off-the-shelf model considered was the OME/UR made by Aerojet. This model provides a maximum nominal thrust of 2.67 X 10^4 N and

a specific impulse of 340 seconds. The mass of each OME/UR engine is 90.72 kg [Wertz].

8.5.2 Propellant feed system

The propellant feed system for attitude control is a blowdown pressurization scheme as shown in Figure 8.5.1.

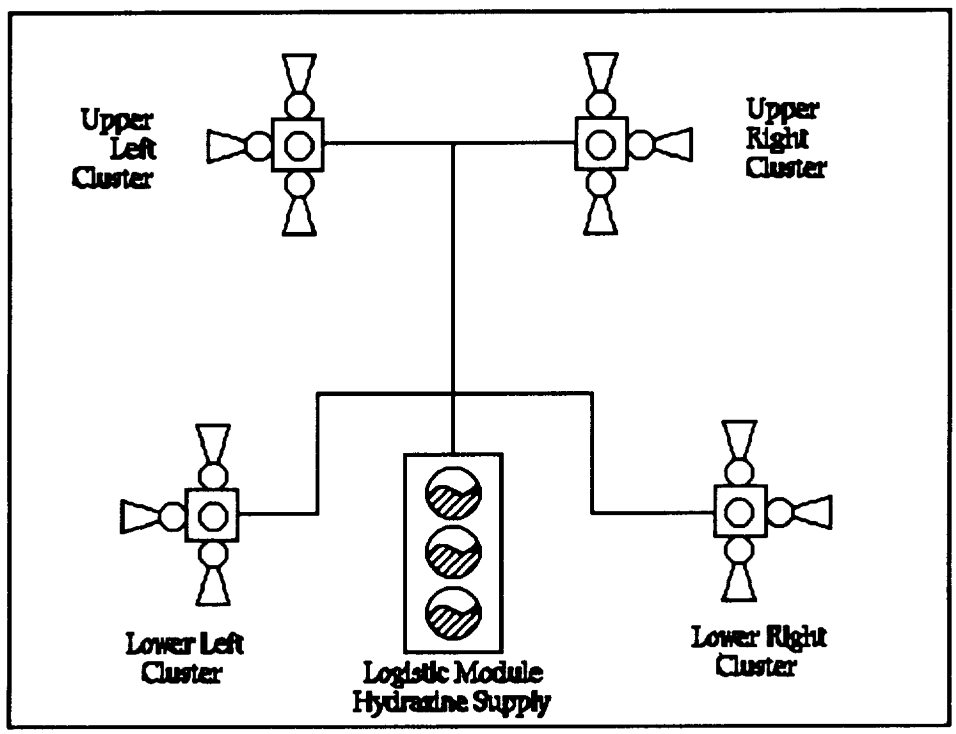

Figure 8.5.1. Blowdown hydrazine thruster system for attitude control.

For the reboost thrusters, each one can have a feed system such as that shown in Figure 8.5.2.

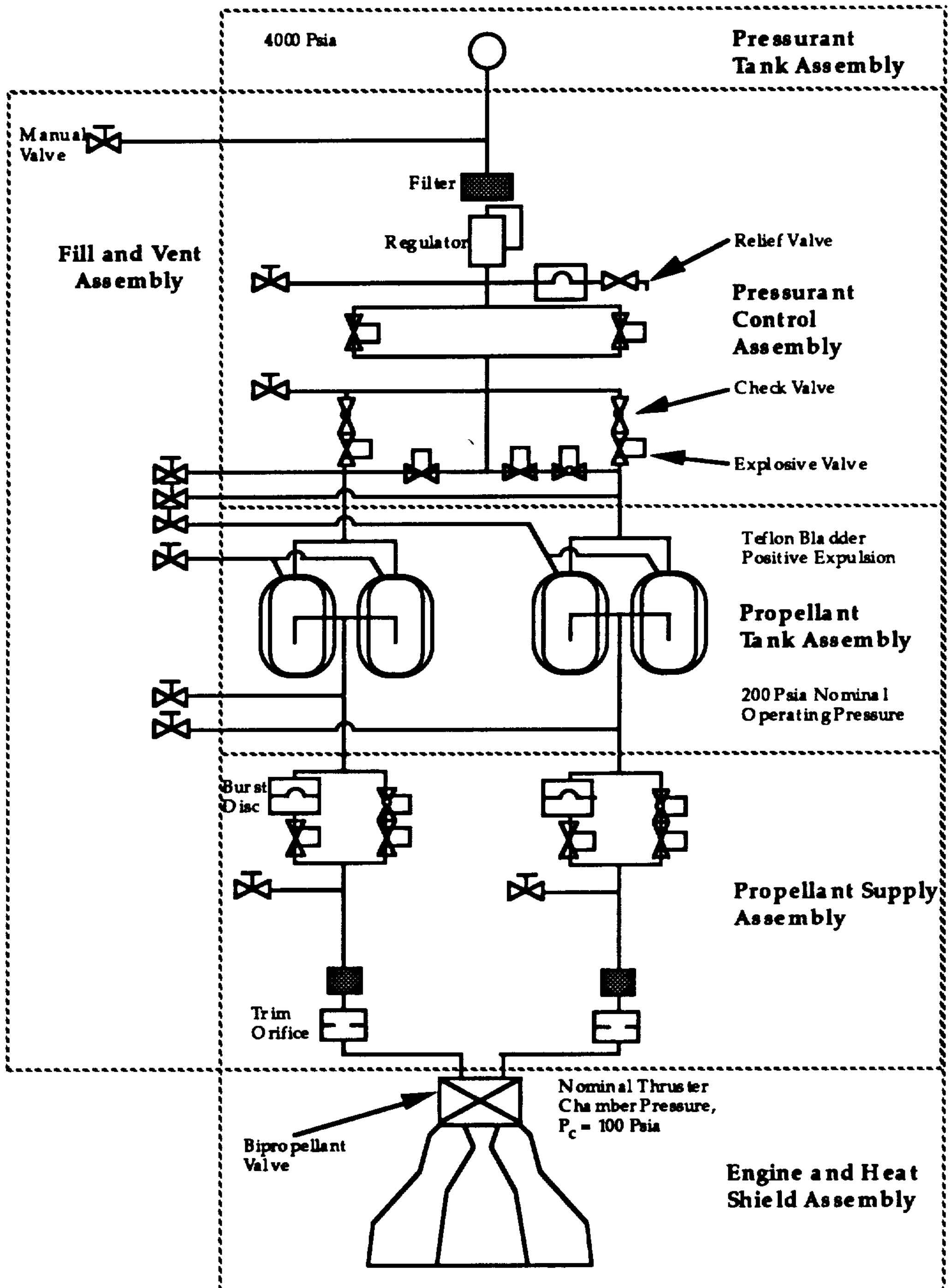

Figure 8.5.2. Pressure-Fed Propulsion System for a N_2O_4/MMH Reboost Thruster [Wertz].

8.6 Future Work

Some problems in designing the propulsion system for the SHARC station, were encountered. Due to time constraints, some areas were not researched in detail, including:

- Propellant storage and feed systems for attitude thrusters
- Ion engines or arcjets
- Propulsion during docking/berthing

Although section 8.5.2 presents two propellant feed systems, these are only a few of the options available. A further analysis of how the feed system is incorporated into SHARC should be done to narrow the choice of feed systems available.

At the time this report was completed, information on ion engines and arcjets was not available. In the future, research could be done on how SHARC can benefit from them. With the acquisition of an ion engine by the Jet Propulsion Laboratory, important information.

As for docking/berthing, no requirements were set. A method for controlling attitude during docking/berthing should be researched and implemented.

9.0 Communications

9.1 System Objectives

Designing a communications system requires a knowledge of the tasks that SHARC will execute, such as teleoperation of machinery, telemetry, tracking, and docking operations. Several things must be determined before a communications system can be chosen, but ultimately the communications group was responsible for determining the system's size and power requirements for integration with the rest of SHARC. Here is a preliminary look at some of the objectives and requirements that the SHARC communications subsystem must fulfill:

- Continuous voice contact with Houston.
- Audio visual contact with Houston.
- Continuous contact for telemetry, tracking and command, GNC, and EC/LS shutdown.
- Manual control of upcoming payloads from various unmanned launch vehicles.

9.2 System Requirements

Once the objectives were completely defined and the necessary sensors were chosen, the communications subsystem group selected the data rates that SHARC needed. From that point there were four main items to decide upon:

- Frequency spectrum
- Arrangement for continuous coverage with minimum delays
- Antenna size and transmitter power
- Link design

Some other requirements that were defined include the internal noise, accuracy/redundancy, atmospheric and rain attenuation, and thermal constraints.

9.3 Communication System Design

The communications group was separated into two different systems, a local system and a space to ground system. Both communications systems will be modeled after Space Station Freedom, since SSF offers the best approximation of the data rates that will be used on SHARC.

9.3.1 Local System

The local communication system will use an optical network. It will consist of the many on-board sensors involved with telemetry, ECLS, and the robotic operations. Many of the components for optical communications have already been designed and produced, but network integration does not exist. Using current technology, an optical network can handle data rates of 10 Mbps (megabits per second), but a point to point fiber optic connection can increase the data rate up to 100 Mbps.

Laser communications have the advantage of requiring less power and mass, greater reliability, and the capability of meeting the needs of communication systems as they expand. Studies show that the use of lasers in communication networks will have the capability of increasing data rates to 500 Mbps. The use of lasers may also extend to several data link designs.

9.3.2 Space to Ground System

The space to ground communication system will be required to downlink two virtual channels having a data rate of 150 Mbps and an uplink of 25 Mbps. The need for such a high data rate comes from the objective of controlling some of SHARC's functions from the ground. To maintain continuous contact with Houston, this system can be integrated with the Tracking and Data Relay Satellite System (TDRSS). TDRSS will then link up with the Data Interface Facility (DIF) in White Sands, which separates the channels and gets them to the appropriate user. The power requirements for this system are found in section 6.3.1. We will use a center-feed parabolic reflector antenna design having a mass of 4.7 kg and a diameter of 1.7 meters. The location of this antenna can be found in Figure 1.1

The specific frequency will be assigned by the FCC, but it will be in the range of two gigahertz, in order to minimize atmospheric effects on the broadcast signal.

10.0 Thermal Control

10.1 Waste Heat Estimate

A simple energy balance method was used to estimate the waste heat generated by the station. Heat sources are set equal to heat sinks, which defines an equilibrium temperature for the station. Using this temperature for the station trusswork, we created another TK model of how much energy must be removed from the modules to lower their average temperature to 21°C, the optimal habitable temperature.

Four heat sources were considered for this analysis:

- Direct solar radiation
- Solar radiation reflected from the Earth
- Earth blackbody radiation
- Internal energy generation

The last term includes the life support equipment, crew memebers, power line losses, inefficiencies in robots and other equipment, and other sources of waste heat. The two heat sinks considered were SHARC blackbody radiation and fluid-loop radiators. These terms are defined in Appendix F.

10.2 Radiator Panel Sizing

After running the TK model for six different orbital positions, we determined that a peak load of 60 kW of heat would have to be dissipated through the radiator panels. Based on a Freon-12 working fluid, we calculated the necessary radiator panel area. The fluid was assumed to enter the radiator at a temperature of 32.2°C (90°F), the working temperature of current Freedom designs. Based on this, we estimate a 35' x 20' panel will be required. We also

assumed that the surface of the panel would be coated with silverized Teflon, in order to reduce the amount of incoming radiation absorbed. The panel sizing algorithm is also shown in Appendix F.

10.3 Future Work

Although this provides an accurate first estimate of thermal requirements, future models should be run with the following factors considered:

- More points in the orbit to gether more data on station thermal loads
- Conduction between various station elements
- A more detailed model of internal heat generation
- Seperating current station elements into smaller sub-elements for more detailed analysis
- Changing surface thermal characteristics during station lifetime
- Dynamic thermal loading characteristics
- Inefficiencies in the radiator system

11.0 Project Management

11.1 SHARC Cost Analysis

The current fiscal year budget for the Space Station Freedom is approximately $2.2 billion dollars. SHARC will utilize a large percentage of technology from Space Station Freedom; therefore we estimated a yearly budget for design, analysis and manufacturing of only $1.35 billion dollars, for a total of $8 billion dollars to complete by 1998. These costs are based on current dollar values, ignoring such factors as inflation and future values.

The launch budget seems to be a significant contribution to the final cost. Each launch costs an estimated $325 million dollars for either the Space Shuttle or a Titan IV. Using a maximum payload of the shuttle and the Titan of 20,000 kg and 13,000 kg respectively. Based on these figures, we took an average payload of 16,500 kg to calculate the number of launches that will be needed to complete construction of SHARC. This resulted in a calculation of 18 launches over a period of one and a half years. The total launch cost comes to $5.85 billion dollars.

The operation cost takes into consideration replacement of solar panels, batteries, consumables, and propellant. The ground support is estimated to cost $250 million dollars annually. Combining the above maintenance, launch, and support costs, the yearly operation of SHARC will cost about $1.25 billion dollars.

Table: 11.1.1 Total Cost Summary for SHARC

Design, Analysis,and Manufacturing	$8 billion
Launch Cost	$5.85 billion
Operation and Maintenance Cost	$1.25 billion

11.2 Subgroup Organization

Each of the subsystems had its own design team. As the project continues, new teams were formed to design the less critical subsystems while old teams were disbanded as the more immediate work was finished. All the engineers were serving in several subgroups at once to ease integration of each subsystem into the overall design.

There are total of eight subsystem design groups: Attitude/Orbit, Communications, Crew/Life Support, GNC/Reboost, Power Supply, Robotics, Structures/Storage, and Thermal Control. Positioned over all subsytems is the Integration Group, which is in charge of resolving all engineering conflicts between subsystem design groups. Group organization is shown in Table 11.2.1.

Table 11.2.1: Group Organization

Subgroup	Team Leader	Team Members
Attitude/Orbit	D. Hoetger	T. Colangelo, A. Kuo, L. Marcus, P. Tran
Communications	T. Colangelo	A. Kuo, P. Tran
Crew/Life Support	G. Wildgrube	M. Lo, C. Tutt, C. Wassmuth
GNC/Reboost	P. Tran	D. Hoetger, G. Wildgrube
Integration	T. Colangelo	A. Kuo, L. Marcus, C. Tutt
Power Supply	A. Kuo	M. Lo, D. Hoetger, M. Lo
Robotics	M. Lo	L. Marcus, C. Wassmuth
Structures/Storage	C. Wassmuth	T. Colangelo, C. Tutt, G. Wildgrube
Thermal Control	C. Tutt	D. Hoetger, L. Marcus

11.3 Project Schedule

The project proceeded according to schedule with only minor delays. The completed schedule is shown in Figure 11.3.1.

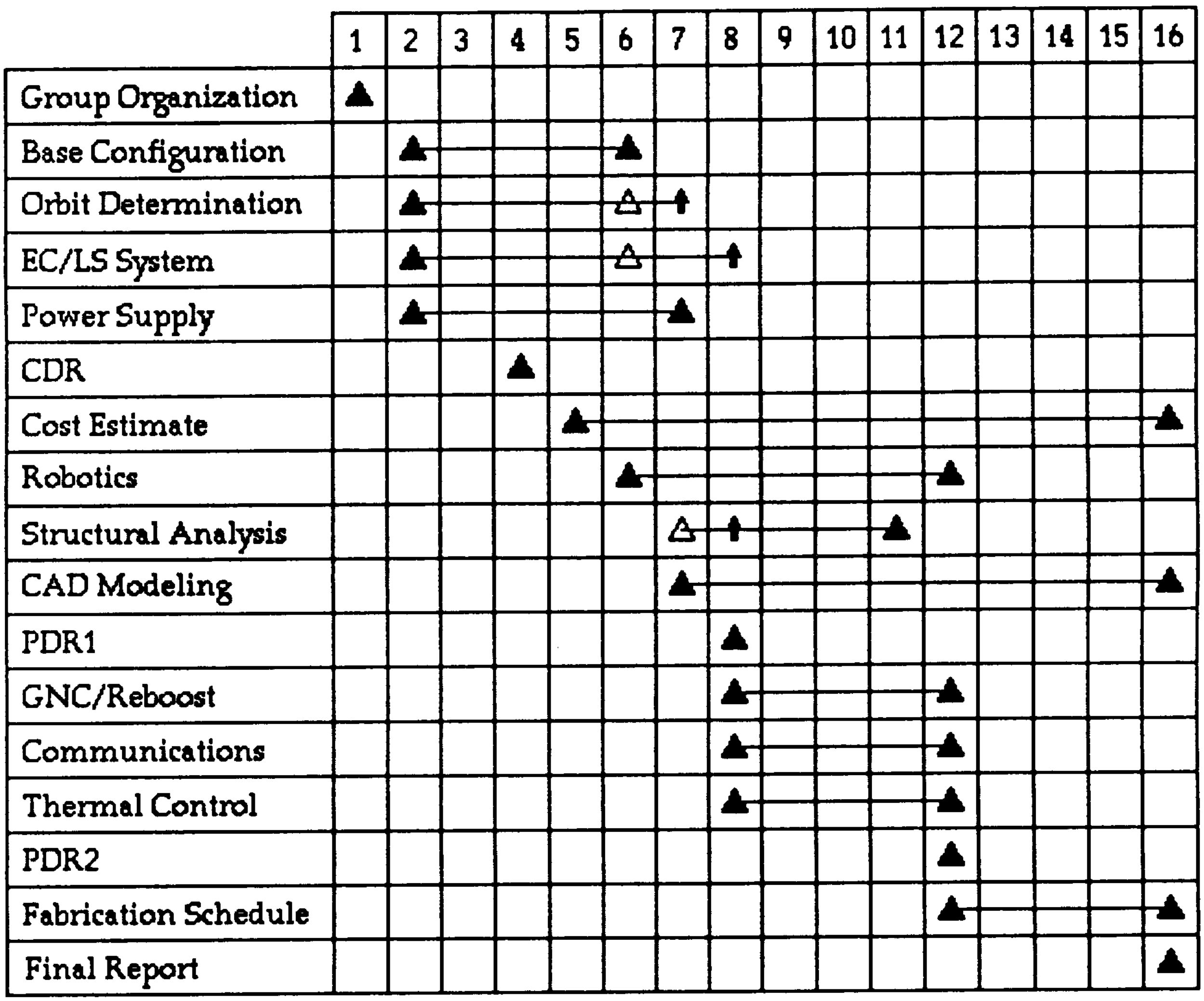

Figure 11.3.1 SHARC Project Timeline

11.4 Personnel Budget

When we organized this project, we predicted an average 16-hour work week for the engineers and a 20-hour work week for the upper management. This gives the projected salaries shown in Table 11.4.1. The actual salaries, based on employee time cards, are also given. This shows that the SHARC project is currently $786 under budget.

Since five hours of consulting work per week were regularly scheduled and material costs were exactly as expected, the salary savings also represents the total contract savings.

Table 11.4.1 Total Personnel Costs

Team Member	Expected Salary	Actual Salary
T. Colangelo	$8,000	$8,555.55
L. Marcus	$7,040	$6,961.78
C. Tutt	$7,040	$7,665.78
D. Hoetger	$4,352	$4,049.78
A. Kuo	$4,352	$4,246.22
M. Lo	$4,352	$4,412.44
P. Tran	$4,352	$4,140.44
C. Wassmuth	$4,352	$4,034.67
G. Wildgrube	$4,352	$3,338.67
Total	$48,192	$47,405.33

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Appendix A: Orbit Analysis Routines

A.1 Original ΔV and TOF program

This TK! Solver model calculated the Δv's and time of flight for a Hohmann transfer to a given orbit as a function of altitude. The program was written by Medha Date at the University of Texas.

Rules:

```
Et=-398601.2/(r1+r2)
v1=(2*(398601.2/r1+Et))^.5
vcs1=(398601.2/r1)^.5
v2=(2*(398601.2/r2+Et))^.5
vcs2=(398601.2/r2)^.5
dv1=v1-vcs1
dv2=v2-vcs2
dvtot=dv1+dv2
tof=pi()*((r1+r2)/2)^1.5/(398601.2)^.5
r1=h1+6378.145
r2=h2+6378.145
```

Variables:

St	input	Name	Output	Unit	Comment
L		Et	0		
L		r1	0		
L		r2	0		
L		v1	0		
L		vcs1	0		
L		v2	0		
L		vcs2	0		
L		dv1	0		
L		dv2	0		
L		dvtot	0		
L		tof	0	hr	
L	0	h1			
L	0	h2			

A.2 ΔV, TOF, and Launch Window program

The TK! Solver model listed in section A.1 was modified by Debora Hoetger for ISS to include synodic period and launch window calculations.

Rules:

```
*T=Tsec/86400
* Et=-398601.2/(r1+r2)
* v1=(2*(398601.2/r1+Et))^.5
* vcs1=(398601.2/r1)^.5
* v2=(2*(398601.2/r2+Et))^.5
* vcs2=(398601.2/r2)^.5
* dv1=v1-vcs1
* dv2=v2-vcs2
* dvtot=dv1+dv2
* tof=pi()*((r1+r2)/2)^1.5/(398601.2)^.5
* r1=h1+6378.145
* r2=h2+6378.145
* wcs=(g/r1)^0.5
* wcf=(g/r2)^0.5
* wrel=wcs-wcf
* Tinter=2*pi()/wrel*100000
* Tsec=Tinter/100000
* T=Tsec/86400
```

Variables:

St	input	Name	Output	Unit	Comment
L		r1	0	km	radius 1
L		r2	0	km	radius 2
L		v1	0	km/s	velocity of orbit 1
L		vcs1	0	km/s	vel.circ.sat. 1
L		v2	0	km/s	velocity of orbit 2
L		vcs2	0	km/s	vel.circ.sat. 2
L		dv1	0	km/s	delta v 1

St	input	Name	Output	Unit	Comment
L		dv2	0	km/s	delta v 2
L		dvtot 0		km/s	total delta v
L		tof	0	hr	time of flight
L	0	h1		km	altitude of SHARC
L	0	h2		km	altitude of SSF
		wc			ang. vel
	.00981	g		km/s^2	gravity
L		wcs		0	ang. vel. of SHARC
L		wcf	0		ang. vel. of SSF
L		wel	0		relative ang. vel.
L		T	0	days	Synodic Period
		pi			
L		Tinter	0	sec	intermediate step
L		Tsec	0	sec	synodic period

A.3 Drag and δa/δt program

This program calculates change the initial decay rate of semi-major axis and the drag force for a given altitude. It was written by Debora Hoetger for ISS.

Rules:

*dadt=-(2*vcs1^2*D*h1^2/(398601.2*mass))* vcs2=(398601.2/r2)^.5

* vcs1=(398601.2/r1)^.5

* r1=h1+6378.145

* r2=h2+6378.145

* D=(Cd*0.5*RHO*vcs1^2*AREA)*1000.0

* dadt=-(2*vcs1^2*D*h1^2/(398601.2*mass))

Variables:

St	input	Name	Output	Unit	Comment
L		r1	0	km	dis. from cent. of earth
L		r2	0		
L		vcs1	0	km/sec	orbital velocity
L		vcs2	0		
L	0	h1		km	Alt. to SHARC
L	0	h2		km	Alt. to SSF

St	input	Name	Output	Unit	Comment
L		D	0	N	Drag
	2	Cd			Coeff. of Drag
L	0	RHO			dens. in kg/km^3
L	0	AREA		km^2	area
L		dadt	0	m/sec	change in semi-maj. axis
L	0	mass		Kg	mass of SHARC

A.4 Program DENSITY

This program calculates atmospheric density as a function of altitude for Low Earth Orbit ranges. It was written for ISS by Phillip Tran. The Subroutine DENS76 was originally written by Johnny Kwok of JPL. DENS76 calculates atmospheric density based on the 1976 U. S. Standard Atmosphere.

```
      PROGRAM DENSITY
      REAL*8 H,DENS,DH
      INTEGER*2 I
      OPEN(10,FILE="DENSITY.OUT")
      H = 300.D0
      DH = 1.D0
      DO 100 I=1,100
      CALL DENS76(H,DENS)
      WRITE(10,1000)H,DENS
      H = H + DH
 100  CONTINUE
 1000 FORMAT(E10.3,E15.8)
      STOP
      END
```

OUTPUT:

Altitude	Density	Altitude	Density
0.300E+03	0.19160000E-01	0.344E+03	0.78690000E-02
0.301E+03	0.18760000E-01	0.345E+03	0.77205000E-02
0.302E+03	0.18360000E-01	0.346E+03	0.75720000E-02
0.303E+03	0.17980000E-01	0.347E+03	0.74295000E-02
0.304E+03	0.17600000E-01	0.348E+03	0.72870000E-02
0.305E+03	0.17240000E-01	0.349E+03	0.71505000E-02
0.306E+03	0.16880000E-01	0.350E+03	0.70140000E-02
0.307E+03	0.16530000E-01	0.351E+03	0.68825000E-02
0.308E+03	0.16180000E-01	0.352E+03	0.67510000E-02
0.310E+03	0.15520000E-01	0.353E+03	0.66255000E-02
0.311E+03	0.15205000E-01	0.354E+03	0.65000000E-02
0.312E+03	0.14890000E-01	0.356E+03	0.62590000E-02
0.313E+03	0.14590000E-01	0.357E+03	0.61430000E-02
0.314E+03	0.14290000E-01	0.358E+03	0.60270000E-02
0.315E+03	0.14005000E-01	0.359E+03	0.59160000E-02
0.316E+03	0.13720000E-01	0.360E+03	0.58050000E-02
0.317E+03	0.13445000E-01	0.361E+03	0.56985000E-02
0.318E+03	0.13170000E-01	0.362E+03	0.55920000E-02
0.319E+03	0.12905000E-01	0.363E+03	0.54895000E-02
0.320E+03	0.12640000E-01	0.364E+03	0.53870000E-02
0.321E+03	0.12390000E-01	0.365E+03	0.52885000E-02
0.322E+03	0.12140000E-01	0.366E+03	0.51900000E-02
0.323E+03	0.11900000E-01	0.367E+03	0.50955000E-02
0.324E+03	0.11660000E-01	0.368E+03	0.50010000E-02
0.325E+03	0.11435000E-01	0.369E+03	0.49105000E-02
0.326E+03	0.11210000E-01	0.370E+03	0.48200000E-02
0.327E+03	0.10990000E-01	0.371E+03	0.47325000E-02
0.328E+03	0.10770000E-01	0.372E+03	0.46450000E-02
0.329E+03	0.10560000E-01	0.373E+03	0.45615000E-02
0.330E+03	0.10350000E-01	0.374E+03	0.44780000E-02
0.331E+03	0.10148000E-01	0.375E+03	0.43970000E-02
0.332E+03	0.99460000E-02	0.376E+03	0.43160000E-02
0.333E+03	0.97535000E-02	0.377E+03	0.42390000E-02
0.334E+03	0.95610000E-02	0.378E+03	0.41620000E-02
0.335E+03	0.93770000E-02	0.379E+03	0.40875000E-02
0.336E+03	0.91930000E-02	0.380E+03	0.40130000E-02
0.337E+03	0.90170000E-02	0.381E+03	0.39415000E-02
0.338E+03	0.88410000E-02	0.382E+03	0.38700000E-02
0.339E+03	0.86720000E-02	0.383E+03	0.38010000E-02
0.340E+03	0.85030000E-02	0.384E+03	0.37320000E-02
0.341E+03	0.83410000E-02	0.385E+03	0.36655000E-02
0.342E+03	0.81790000E-02	0.386E+03	0.35990000E-02
0.343E+03	0.80240000E-02	0.387E+03	0.35355000E-02

Appendix B - Station Configuration Design Matrix

B.1 Matrix Criteria

In order to determine which station concept best suited our needs, we had to come up with some criteria to base our decision on. The first six criteria we used were the AIAA/LORAL requirements from the RFP:

- Ability to assemble three vehicles at once
- Ease of access to parts storage areas from the assembly bays
- Ability to minimize EVA during assembly operations
- Ease of reboosting
- Ability to dock with a wide range of vehicles
- Ease of initial station deployment and assembly operations

Each of these criteria was assigned a weighting factor of 10. After this, the integration team came up with ten other criteria which would help in selecting the best option. They are listed below with the weighting factor in parentheses:

- Material costs for the station (10)
- Projected crew safety in a catastrophic failure (10)
- Projected construction time before operations can begin (8)
- Concept originality (7)
- Projected drag and orbital lifetime (6)
- Ease of power supply mounting and power distribution (6)
- Ease of attitude mounting and station attitude control (5)
- Ability to expand for future operations (5)
- Ease of robot mounting for assembly procedures (5)

B.2 Configuration Concepts

Twelve possible station configurations were considered by the ISS design group. Each one was run through the design matrix by each team member and then the scores were averaged. The results are shown in Table B.2.1 below. Based on this, the two Hammerhead designs were chosen for SHARC as primary and alternate station configuration. Each configuration is discussed in more detail below.

B.2.1 Flagpole

The Flagpole station was an attempt to design a station with an absolute minimum of in-orbit construction. This station has a long keel, to which the command and habitation modules are attached. The bays are placed axially along the station. Power is provided by solar arrays at one end of the keel. The biggest problem with this type of structure is achieving adequate structural stiffness.

B.2.2 Pipe

In the Pipe design, the station modules are laid end-to-end and enclosed by trusswork to increase stiffness. Shuttle docking and vehicle assembly could occur all around the station. The solar panels lie flat on the end of the station, along with fuel storage. This design was an attempt to minimize atmospheric drag.

B.2.3 Twin Boom

In this design, the modules form the central structure, with assembly operations going on in the enclosed square. The solar panels are extended on one boom to allow them freedom of movement while the fuel is stored on the

other boom away from all the other station elements. This maximizes crew safety and reduces the complexity of orbital construction.

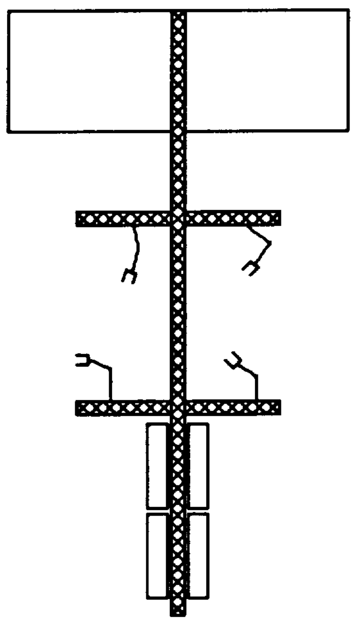


Figure B.2.1 Flagpole Design

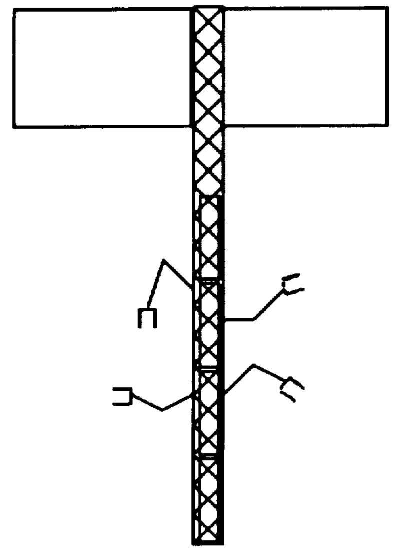


Figure B.2.2 Pipe Design

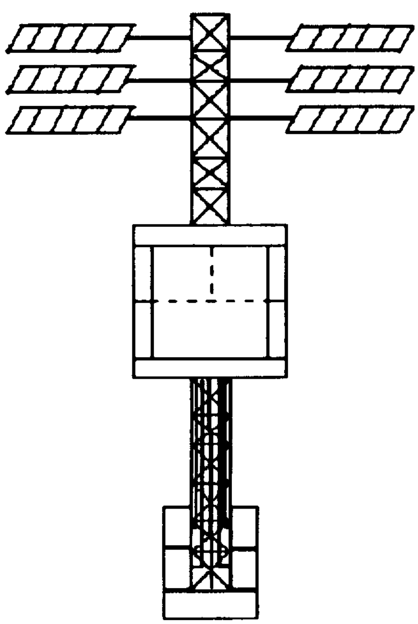


Figure B.2.3 Twin Boom Design

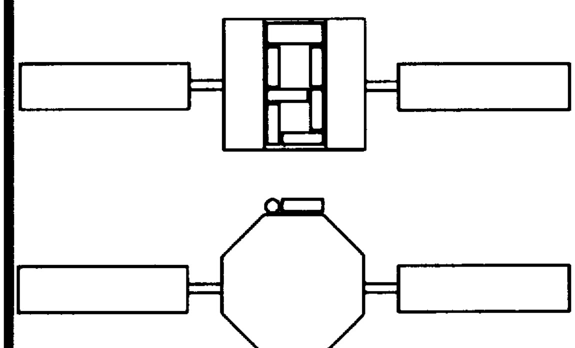


Figure B.2.4 Octagon Design

B.2.4 Octagon

The main focus of this design was ease of assembly operations. All three vehicles would be assembled in the large (90' dia.) central bay. Parts storage and the pressurized garage would open directly into the assembly bay. Robots could be mounted on tracks running down the interior of the bay. This design would require large amounts of orbital assembly before operations could begin.

B.2.5 Octagon II

This design was almost identical to the Octagon. The solar panels were moved to help simplify attitude control and also reduce orbital drag. Like the original design, it required large amounts of orbital construction work before operations could begin.

B.2.6 Cross

This was a quite elaborate design and the first to use a track system for fuel storage. It allowed the fuel to be moved away from any vital station components. The simple docking facilities could accommodate even the largest vehicles. The major drawbacks were probable flexibility problems and the difficulty of assembling it in orbit.

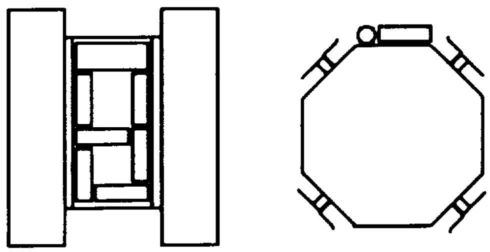


Figure B.2.5 Octagon II Design

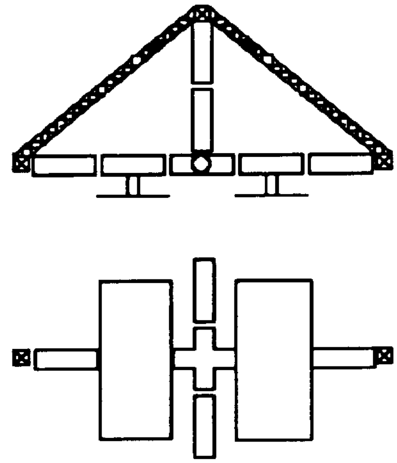
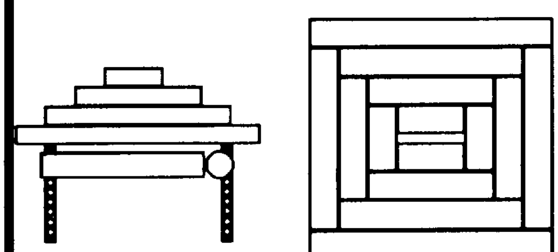


Figure B.2.6 Cross Design



B.2.7 House Design

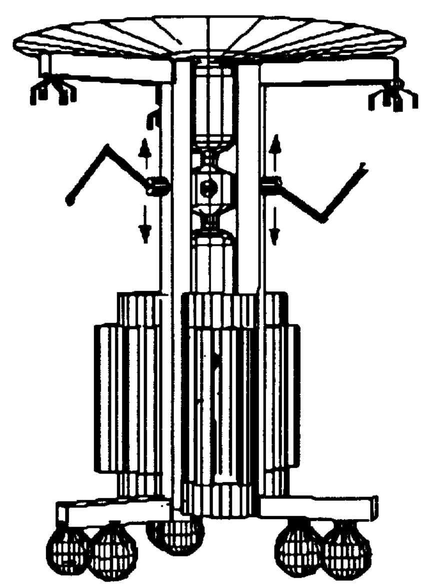


Figure B.2.8 Tripod Design

B.2.7 House Design

The House was designed to prevent thermal loading on the truss structure and the vehicles being assembled, while also easing the thermal control problems for the habitation modules. The solar array would cover the entire top of the station in a "roof" configuration. Vehicle assembly would take place below this roof, with all parts being stored inside. Although the connections for the solar panels would probably be quite intricate, the rest of the station could be put together relatively easily.

B.2.8 Tripod

In this configuration, the modules form the keel of the station. Trusswork surrounds these modules to increase structural stiffness. The large cylinders on the end are storage bays and the spheres are fuel storage tanks. This concept had low drag, but crew safety was a major concern.

B.2.9 Hammerhead I

This is the alternate design for SHARC. Although it received the highest rating, concerns about subsystem mounting prompted a switch to Hammerhead II for the primary design. This design has a small enclosed bay and two larger exposed bays. Movable side trusses allow the largest bay to expand to service large vehicles. The limited assembly requirements allow operations to begin very quickly.

B.2.10 Hammerhead II

This is the station design chosen for SHARC. The full details of this structure are given in Section 3.0.

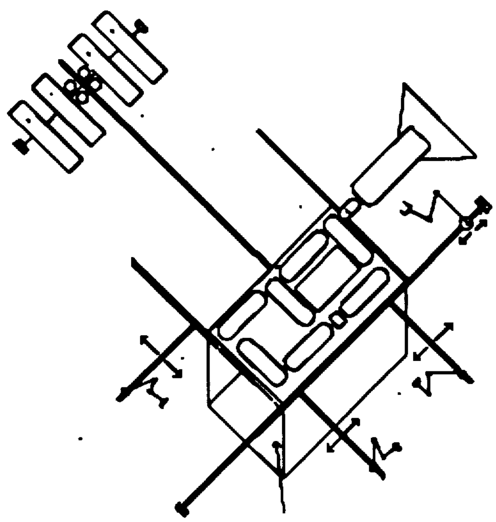


Figure B.2.9 Hammerhead I Design

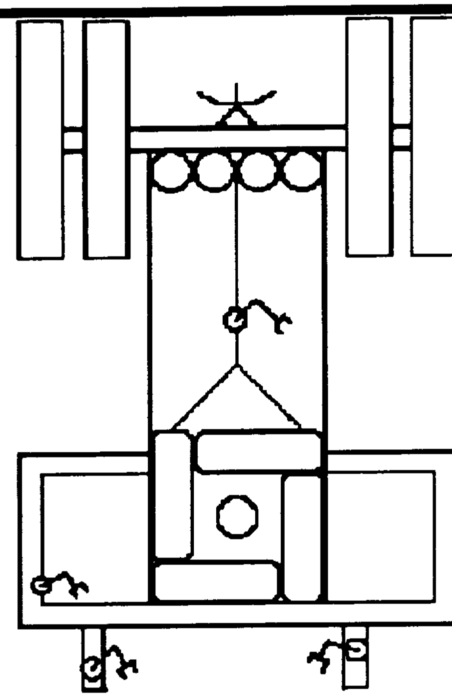


Figure B.2.10 Hammerhead II Design

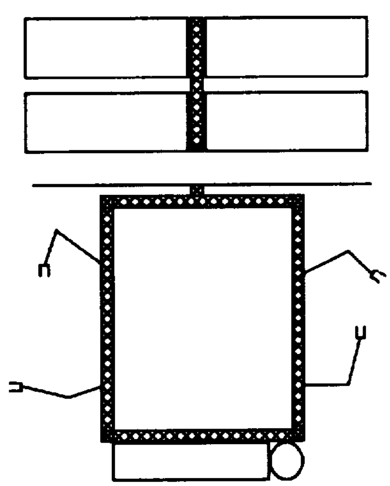


Figure B.2.11 Mushroom Design

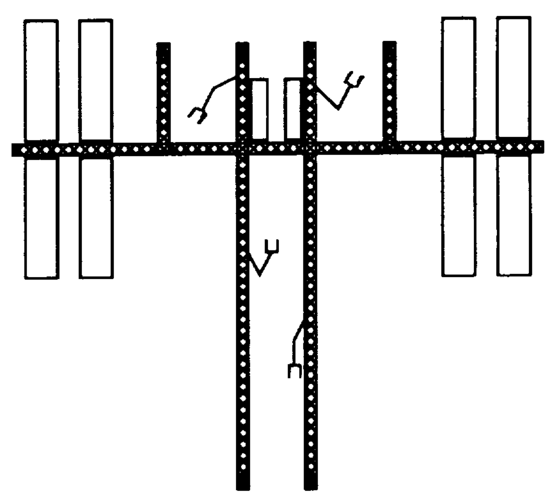


Figure B.2.12 Cannon Design

B.2.11 Mushroom Design

The Mushroom station is an attempt to minimize drag by making the station as compact as possible. The solar panels are arranged on top, forming the "cap" of the Mushroom. The other station elements will be permanently shaded, easing thermal control for the modules and preventing cyclic thermal loads in the structure. The "stalk" consists of trusswork enclosing storage areas and modules. Major assembly would occur adjacent to the stalk.

B.2.12 Cannon

This design maximized expandability. The station capabilities could easily be increased to accommodate vehicles of any size. In addition, the attitude control jets would have large moment arms to maximize control authority. The major concern about this idea was inadequate structural stiffness.

Table B.2.1: Configuration Design Matrix Ratings

Hammerhead I	920
Hammerhead II	919
Mushroom	833
Pipe	797
Twin Boom	788
Tripod	781
Octagon II	775
Flagpole	770
Cannon	752
House	748
Octagon	742
Cross	666

Appendix C - Life Support Algorithms

C.1 University of Texas FORTRAN Code

This code was developed by James Dugan and Nathan Nottke as part of the class requirements in ASE 396, Space Systems Design. This code was modified by Chris Tutt of ISS to speed processing. The source code and three sample runs are shown below.

C.1.1 Source Code

```
************************************
* Program ECLSS                    *
* This program sizes an ECLSS      *
* subsystem subject to the user's  *
* choice of the degree of closure  *
*                                  *
* The following technologies are   *
* considered:                      *
*                                  *
* 1. Open System:                  *
*   - LiOH for CO2 removal         *
*   - Storage of all water,oxygen  *
*                                  *
* 2. Partially Closed System:      *
*   - EDC for CO2 removal          *
*   - Storage of water for CO2     *
*     reduction and urine/waste    *
*   - Storage of all oxygen        *
*   - MF for wash water recovery   *
*                                  *
* 3. Closed System:                *
*   - EDC for CO2 removal          *
*   - SAB reactor to collect CO2   *
*   - SF Electrolysis for O2       *
*   - VCD for urine/waste water    *
*   - MF for wash water recovery   *
*                                  *
* See the subsystem manual for a   *
* description of each of these     *
* technologies.                    *
```

```
***********************************
* 1. Determine the size of the    *
*    mission to be supported      *
***********************************
      Program ECLSS
      Implicit Real(A-H,O-Z)
      Write(9,*)'Input the number of crew to be supported:'
      Read(9,*)C
      Write (9,*)' '
          Write(9,*)'Input  the  number  of  days  they  must  be
supported:'
      Read(9,*)T
      Write(9,*)' '
      Write(9,*)'Input the type of ECLSS system desired.'
      Write(9,*)'Type the number next to your choice:'
    1 Write(9,*)'   1. Open'
      Write(9,*)'   2. Partially closed'
      Write(9,*)'   3. Closed'
      Read(9,*)IE
      ErrFlag = 0
***********************************
* 2. Create first estimate of     *
*    subsystem size and supply    *
*    requirements                 *
***********************************
      Select Case (IE)
      Case(1)
       PR=C*0.195
       WH=C*0.21
       SCM=((C*3131.0775+83.61)/90)*T
       SCV=((C*3.241)/90.0)*T
       HWM=C*76.1-83.61
       HWV=C*7.4165
     Case(2)
       PR=C*0.2475
       WH=C*0.3225
       SCM=((C*617.6345+83.61)/90.0)*T
       SCV=((C*0.8585)/90.0)*T
       HWM=C*87.018-83.61
       HWV=C*1.6095
     Case(3)
       PR=C*0.4475
       WH=C*0.390
```

```
        SCM=((C*152.4826+83.61)/90.0)*T
        SCV=((C*0.42202)/90.0)*T
        HWM=C*108.314-83.61
        HWV=C*0.73353
      Case Default
        ErrFlag = 1
       End Select
       If(ErrFlag.Eq.1) Go to 1
       TotSM=SCM+HWM
       TotSV=SCV+HWV
       PI=PR*359.0
       HI=WH*109.0
**********************************
* 3. Print results               *
**********************************
       Do 2 I=1,10
         Write(9,*)' '
     2 Continue
     3 Format(1X,A,F10.3)
     4 Format(1X,A,F3.0)
       Write(6,*)'ECLSS System Estimate'
       Write(6,*)'---------------------'
       Write(6,*)' '
       Write(6,4)'Designed Crew Size                     ',C
       Write(6,4)'Designed Mission Duration (days)       ',T
       Write(6,*)' '
       Write(6,3)'Power Required (kW)                    ',PR
       Write(6,3)'Waste Heat Generated (kW)              ',WH
       Write(6,*)' '
       Write(6,3)'Mass of Spares/Consumables (kg)        ',SCM
       Write(6,3)'Mass of System Hardware (kg)           ',HWM
       Write(6,3)'Total System Mass (kg)                 ',TotSM
       Write(6,*)' '
       Write(6,3)'Volume of Spares/Consumables (m^3) ',SCV
       Write(6,3)'Volume of System Hardware (m^3)        ',HWV
       Write(6,3)'Total System Volume (m^3)              ',TotSV
       Write(6,*)' '
       Write(6,3)'Power Impact Penalty (kg)              ',PI
       Write(6,3)'Waste Heat Impact Penalty (kg)         ',HI
       End
```

C.1.2 Sample Runs

The crew scenario for SHARC, eight crew for 35 days, was run through this program to get the system size estimates:

A) Open Loop System

```
ECLSS System Estimate
---------------------

Designed Crew Size                          8.
Designed Mission Duration (days)           35.

Power Required (kW)                         1.560
Waste Heat Generated (kW)                   1.680

Mass of Spares/Consumables (kg)          9773.645
Mass of System Hardware (kg)              525.190
Total System Mass (kg)                  10298.833

Volume of Spares/Consumables (m^3)         10.083
Volume of System Hardware (m^3)            59.332
Total System Volume (m^3)                  69.415

Power Impact Penalty (kg)                 560.040
Waste Heat Impact Penalty (kg)            183.120
```

B) Partially Closed Loop System

```
ECLSS System Estimate
---------------------

Designed Crew Size                          8.
Designed Mission Duration (days)           35.

Power Required (kW)                         1.560
Waste Heat Generated (kW)                   1.680

Mass of Spares/Consumables (kg)           1954.045
Mass of System Hardware (kg)              612.534
Total System Mass (kg)                   2566.579
```

Volume of Spares/Consumables (m^3)	2.671
Volume of System Hardware (m^3)	12.876
Total System Volume (m^3)	15.547
Power Impact Penalty (kg)	710.820
Waste Heat Impact Penalty (kg)	281.220

C) Closed Loop System

ECLSS System Estimate

Designed Crew Size	8.
Designed Mission Duration (days)	35.
Power Required (kW)	3.580
Waste Heat Generated (kW)	3.120
Mass of Spares/Consumables (kg)	506.905
Mass of System Hardware (kg)	782.902
Total System Mass (kg)	1289.807
Volume of Spares/Consumables (m^3)	1.313
Volume of System Hardware (m^3)	5.868
Total System Volume (m^3)	7.181
Power Impact Penalty (kg)	1285.220
Waste Heat Impact Penalty (kg)	340.080

C.2 Woodcock Algorithm

The algorithm used for these calculations can be found in Appendix C of Woodcock's book *Space Stations and Platforms.* It was designed for sizing the life support system on Space Station Freedom. Most of the assumptions were carried over and used for the SHARC estimate.

Step 1: Food Consumption

The first thing we had to determine was an average metabolic rate for the crew. NASA uses 136 W for IVA and 400 W for EVA. Woodcock assumes that Freedom will have 16 hours of EVA per crewmember per week. Since one of our requirements is to minimize EVA through telerobotics, we assumed 8 EVA hours for SHARC. Based on this, we can calculate crew average metabolic rate:

$$AMR = \frac{(136\text{ W})(160\text{ h}) + (400\text{W})(8\text{ h})}{168\text{ h}} = 148.57\text{ kW} = 3066\text{ Kcal/day}$$

Food is modeled as a composite molecule, $CH_{1.71}O_{0.54}N_{0.03}J$, where C, H, O, and N have their standard chemical meanings, and J represents the undigestable portion of the food. This molecule decomposes in the body through the following reaction:

$$CH_{1.71}O_{0.54}N_{0.03}J + 1.2925O_2 \dashrightarrow CO_2 + 0.585\,H_2O + 0.54OH + 0.015N_2 + J$$

Decomposition to CO_2 produces 94.385 KCal/gmole

Deomposition to H_2O produces 57.8 KCal/gmole

The other reactions are assumed to be isothermal to account for the energy used by the body to process the food. Based on this, we can calculate the food heat content:

$$FHC=\left(\frac{1\text{ mole }CO_2}{\text{mole food}} * \frac{94.385\text{ KCal}}{\text{gmole }CO_2}\right)+\left(\frac{0.585\text{ mole }H_2O}{\text{mole food}} * \frac{57.8\text{ KCal}}{\text{gmole }H_2O}\right)=\frac{128.2\text{ KCal}}{\text{gmole food}}$$

One gmole of food weighs:

$$\frac{12\text{g C}}{\text{gmole}} + 1.71\left(\frac{1\text{g H}}{\text{gmole}}\right) + 0.54\left(\frac{16\text{g O}}{\text{gmole}}\right) + 0.03\left(\frac{14\text{g N}}{\text{gmole}}\right) + \frac{4.82\text{g J}}{\text{gmole}} = 27.62\text{ g}$$

Therefore, the food heat content by weight is:

$$\frac{128.2 \text{ KCal}}{\text{gmole}} * \frac{1 \text{ gmole}}{27.62 \text{ g}} = 4.642 \text{ KCal/g} = 4642 \text{ KCal/kg}$$

Note: The molecular weight of J was determined empirically through Skylab and Shuttle mission reports. Each astronaut must eat enough food to produce his average metabolic rate:

$$\frac{3066 \text{ KCal/day}}{4642 \text{ KCal/kg}} = \frac{0.66 \text{ kg dry food}}{\text{day - crewmember}}$$

Based on this, SHARC will require:

$$\frac{0.66 \text{ kg food}}{\text{day - crewmember}} * 8 \text{ crewmembers} * 35 \text{ days} = 185 \text{ kg dry food per work tour}$$

Note: This is not how much food has to be delivered! This is how much dry food has to be consumed. The actual food weight will be calculated later.

<u>Step 2: Food Preparation and Eating</u>

Woodcock assumes the food will be 40% water:

$$\frac{185 \text{ kg dry food} * 40\%}{60\%} = 123.3 \text{ kg } H_2O$$

The decomposition of the food described above produces a certain amount of water also:

$$185 \text{ kg dry food} * \frac{1 \text{ kgmole}}{27.62 \text{ kg food}} * \frac{0.855 \text{ kgmole } H_2O}{27.62 \text{ kg food}} * \frac{18 \text{ kg } H_2O}{\text{kgmole}} = 103.1 \text{ kg } H_2O$$

Combining these gives the total mass of water contained in the food:

$$123.3 \text{ kg} + 103.1 \text{ kg} = 226.4 \text{ kg } H_2O$$

We can also calculate a wet food mass:

$$185 \text{ kg dry food} + 123.3 \text{ kg } H_2O = 308.3 \text{ kg wet food}$$

Assume 10% of the food delivered is wasted:

$$\frac{10\%(308.3 \text{ kg})}{90\%} = 34.3 \text{ kg wasted food}$$

From this we can calculate the total amount of food that needs to be delivered to SHARC:

$$308.3 \text{ kg food} + 34.3 \text{ kg waste} = \mathbf{342.6 \text{ kg food}}$$

A certain amount of water needs to be used to prepare the food. Woodcock assumes this is 10% of the wet mass:

$$0.1(308.3 \text{ kg}) = 30.8 \text{ kg } H_20 \text{ eaten by astronauts}$$
$$0.1(34.3 \text{ kg}) = 3.4 \text{ kg thrown away with wasted food}$$
$$30.8 \text{ kg} + 3.4 \text{ kg} = 34.2 \text{ kg } H_2O \text{ needed to prepare food}$$

<u>Step 3: Crewmembers</u>

From the food decomposition discussed earlier, we can estimate the amount of dry fecal waste produced by the astronauts:

$$185 \text{ kg dry food} * \frac{1 \text{ kgmole food}}{27.62 \text{ kg food}} * \frac{1 \text{ mole DFW}}{1 \text{ mole food}} * \frac{4.82 \text{ kg DFW}}{1 \text{ mole DFW}} = 32.3 \text{ kg DFW}$$

Assuming feces are 50% water, the astronauts will give off 32.3 kg of H_2O in their feces. Woodcock also has a formula to estimate the amount of sweat and water vapor given off by the astronauts:

$$
\begin{aligned}
\text{Water vapor} &= (\text{Avg. Metabolic Rate} - 75\text{W}) * 0.0359 \\
&= (148.57\text{W} - 75\text{W}) * 0.0359 \\
&= \frac{2.6 \text{ kg}}{\text{day - crewmember}}
\end{aligned}
$$

$$\frac{2.6 \text{ kg}}{\text{day - crewmember}} * 8 \text{ crewmembers} * 35 \text{ days} = 739.5 \text{ kg } H_2O$$

Woodcock also estimates the amount of urine the astronauts will produce:

$$\frac{1.5 \text{ kg urine}}{\text{day - crewmember}} * 8 \text{ crewmembers} * 35 \text{ days} = 420 \text{ kg } H_2O$$

Equating these will tell you how much water the astronauts must drink:

Urine	420 kg
+ Feces	32.3 kg
+ Sweat	739.5 kg
- Water in Food	226.4 kg
- Food Prep Water	38.1 kg
Drinking Water	934.6 kg

Step 4: Cabin Humidity Control

The amount of water vapor released by the crew was found to be 739.5 kg. Assuming that each crewmember takes one shower per day and uses 5 kg of water per shower, we can estimate the amount of water needed for showers:

$$\frac{5 \text{ kg } H_2O}{\text{shower}} * \frac{1 \text{ shower}}{\text{crewmember - day}} * 8 \text{ crewmembers} * 35 \text{ days} = 1400 \text{ kg } H_2O$$

For nominal air and water temperatures, 3.364% of this water is converted to vapor:

$$.03364(1400\text{ kg}) = 47.1\text{ kg } H_2O\text{ vapor}$$

For clothes washing, Woodcock estimates 1 kg of H_2O per crewmember per wash. When the clothes are dried, all this water is converted into vapor:

$$\frac{1\text{ kg } H_2O}{1\text{ day - crewmember}} * 8\text{ crewmembers} * 35\text{ days} = 280\text{ kg } H_2O\text{ vapor}$$

Woodcock then lists the nominal cabin conditions:

Temperature = 20°C
Pressure = 1 atm = 101.325 kPa
Cabin Humidity = 50%
Air composition = 80% N_2, 20% O_2
5 kg nominal air leakage per day
1 airlock cycle every 4 EVA hours
0.2 kg air loss/lock op
H_2O vapor partial density at nominal conditions: 0.0173 kg/m^3
H_2O parital pressure at 100% quality: 2.34 kPa

The first thing we need to calculate is the molecular weight of the air:

$$\text{Air MW} = 80\%\left(\frac{28\text{ kg } N_2}{\text{kgmole air}}\right) + 20\%\left(\frac{32\text{ kg } O_2}{\text{kgmole air}}\right) = \frac{28.8\text{ kg}}{\text{kgmole air}}$$

We also need the partial pressure of the air:

$$\begin{aligned}\text{Air partial pressure} &= \text{Cabin pressure - } H_2O\text{ partial pressure}\\ &= 101.325\text{ kPa} - 50\%(2.34\text{ kPa}) = 101.155\text{ kPa}\end{aligned}$$

From this we can calculate the partial density of the air:

$$\text{Air partial density} = \frac{PM}{RT} = \frac{(101.155\text{ kPa})(28.8\text{ kg air/kgmole})}{(8316)(293\text{ K})} = 1.184\text{ kg/m}^3$$

Using the partial density, we can get the actual density of the cabin air:

Cabin density = Air density + Water vapor density

$$= 1.184 \text{ kg/m}^3 + 50\%(0.0173 \text{ kg/m}^3) = 1.192 \text{ kg/m}^3$$

We will also need the nominal air density:

$$\text{Air nominal density} = \frac{PM}{RT} = \frac{(101.325 \text{ kPa})(28.8 \text{ kg air/kgmole})}{(8316)(293 \text{ K})} = 1.198 \text{ kg/m}^3$$

Comparing the nominal density with the actual cabin density, we can get the loss factor. This "fudge factor" corrects the air loss calculations for humidity effects:

$$\text{Loss Factor} = \frac{\text{Cabin density}}{\text{Air nominal density}} = \frac{1.192 \text{ kg/m}^3}{1.198 \text{ kg/m}^3} = 0.995$$

Using this loss factor, we can calculate how much air is actually loss during normal operations:

Actual leakage = Nominal leakage * loss factor

$$= \frac{5 \text{ kg}}{\text{day}} * 0.995 = \frac{4.977 \text{ kg}}{\text{day}}$$

$$\frac{4.977\text{kg}}{\text{day}} * 35 \text{ days} = 174.2 \text{ kg air leakage}$$

The air lost through airlock operations can also be estimated using Freedom's assumptions:

$$\text{Airlock loss} = \frac{1 \text{ lock op}}{4 \text{ EVA hrs}} * \frac{8 \text{ EVA hrs}}{\text{astronaut - week}} * 8 \text{ astronauts} * 5 \text{ weeks} * \frac{0.2 \text{ kg}}{\text{lock op}}$$

$$= 16 \text{ kg airlock loss}$$

Total air loss = Module leakage + Airlock losses = 174.2 kg + 16 kg = 190.2 kg

To calculate how exactly what is lost, we need the partial pressures of the various gases:

$$O_2 \text{ partial density} = \frac{32 \text{ kg } O_2/\text{mole}}{28.8 \text{ kg air/mole}} * \frac{1 \text{ mole } O_2}{5 \text{ mole air}} * 1.184 \text{ kg/m}^3 = 0.2631 \text{ kg/m}^3$$

$$N_2 \text{ partial density} = \frac{28 \text{ kg } N_2/\text{mole}}{28.8 \text{ kg air/mole}} * \frac{4 \text{ mole } N_2}{5 \text{ mole air}} * 1.184 \text{ kg/m}^3 = 0.9209 \text{ kg/m}^3$$

$$H_2O \text{ partial density} = 50\%(0.0173 \text{ kg/m}^3) = 0.0087 \text{ kg/m}^3$$

Using these partial densities, we can find out how much of each gas is lost to space:

$$O_2 \text{ loss} = \frac{0.2631 \text{ kg/m}^3}{1.192 \text{ kg/m}^3} * (190.2 \text{ kg air lost}) = 42.0 \text{ kg } O_2 \text{ lost}$$

$$N_2 \text{ loss} = \frac{0.9209 \text{ kg/m}^3}{1.192 \text{ kg/m}^3} * (190.2 \text{ kg air lost}) = \mathbf{146.9 \text{ kg } N_2 \text{ lost}}$$

$$H_2O \text{ loss} = \frac{0.0087 \text{ kg/m}^3}{1.192 \text{ kg/m}^3} * (190.2 \text{ kg air lost}) = 1.4 \text{ kg } H_2O \text{ lost}$$

The nitrogen gas lost must be replaced through stores. The oxygen and water vapor should be able to be replaced by the EC/LS system. From this we can also estimate how much water vapor condensate we can recover:

Shower vapor	47.1 kg
+ Human sweat	739.5 kg
+ Clothes washing	280 kg
- Vapor loss through leakage	1.4 kg
Condensate recovery	1066 kg

5. Wash Water

We have already determined that 700 kg of water are used for clothes washing and 1400 kg are used for showering. The only other wash water needed will be for hand washings:

$$\frac{10 \text{ hand washes}}{\text{crewmember - day}} * \frac{0.1 \text{ kg } H_2O}{\text{hand wash}} * 8 \text{ crewmembers} * 35 \text{ days} = 280 \text{ kg } H_2O$$

Using these numbers, we can calculate total wash water usage:

$$700 \text{ kg} + 1400 \text{ kg} + 280 \text{ kg} = 2380 \text{ kg wash water}$$

6. Waste Water

Sources of waste water are humidity condensate, urine, wash water, and vapor recovered from solid waste. The first three terms have been found earlier, but the last one has to be solved for iteratively. As a first guess, assume it is equal to 120 kg.

$$\begin{aligned}\text{Waste Water Input} &= \text{Condensate + Urine + Wash Water + Recovered Vapor} \\ &= 1066 \text{ kg} + 420 \text{ kg} + 2380 \text{ kg} + 120 \text{ kg} = 3986 \text{ kg}\end{aligned}$$

Woodcock assumes that 95% of the waste water can be purified. The remainder of the water is combined with dirt particles to create a heavy sludge, which is then mixed with the solid waste:

$$\text{Recovered water} = 95\%(3986 \text{ kg}) = 3786.7 \text{ kg}$$
$$\text{Sludge generated} = 3986 \text{ kg} - 3786.7 \text{ kg} = 199.3 \text{ kg}$$

Woodcock also assumes that the sludge is 50% water:

$$\text{Water in sludge} = 50\%(199.3 \text{ kg}) = 99.7 \text{ kg } H_2O$$

We now need to calculate the total water in the solid waste:

- Water in human feces = 32.3 kg
- Water contained in uneaten food = 40%(34.3 kg) = 13.7 kg
- Food prep water in uneaten food = 3.8 kg
- Water in sludge = 99.7 kg

Total water in solid waste = 32.3 kg + 13.7 kg + 3.8 kg + 99.7 kg = 149.5 kg

Current estimates are that 80% of this water can be recovered and treated to *waste* water quality:

Recovered water = 80%(149.5 kg) = 119.6 kg

Another iteration can be performed using 119.6 kg as input instead of 120 kg, but we felt that this was good enough for a first estimate.

6. Potable Water

For our purposes, we assumed that all water on the station would be distilled to potable quality. Based on this, we calculated the amount of water available for electrolysis through the Sabatier reactor:

Waste water recovery	3786.7 kg
- Wash water	2380 kg
- Food prep water	34.3 kg
- Drinking water	927.3 kg
Water for Sabatier	445.1 kg

The Sabatier output will be found through iteration in the next step.

7. Electrolysis and Sabatier

The amount of CO_2 that needs to be removed from the air can be calculated from the food decomposition equation:

$$185 \text{ kg dry food} * \frac{1 \text{ kgmole}}{27.62 \text{ kg food}} * \frac{1 \text{ kgmole } CO_2}{1 \text{ kgmole food}} = 6.698 \text{ kgmoles } CO_2$$

The amount of H_2 available to react with the CO_2 can be calculated from the Sabatier input:

$$445.1 \text{ kg } H_2O * \frac{1 \text{ kgmole}}{18 \text{ kg } H_2O} * \frac{1 \text{ kgmole } H_2}{1 \text{ kgmole } H_2O} = 24.728 \text{ kgmoles } H_2$$

The CO_2 is reacted with H_2O to form CH_4 and O_2 through the following reaction:

$$CO_2 + 2H_2O \dashrightarrow CH_4 + 2O_2$$

$$6.698 \text{ kgmoles } CH_4 * \frac{16 \text{ kg } CH_4}{\text{kgmole}} = \mathbf{107.2 \text{ kg } CH_4}$$

The excess H_2 will form H_2O:

$$24.728 \text{ kgmoles} - 2(6.698) \text{ kgmoles} = 11.332 \text{ kgmoles } H_2O$$

$$11.332 \text{ kgmoles } H_2O * \frac{18 \text{ kg } H_2O}{\text{kgmole}} = 204 \text{ kg } H_2O$$

This 204 kg H_2O can now be reinput into the Sabatier or kept as a reserve for emergencies. Therefore, there is no need to resupply water.

Oxygen generation can also be calculated from the reaction given above.

Total O in Sabatier = 2(6.698 kgmoles CO_2) + 24.728 kgmoles H_2O
= 38.124 kgmoles O

Oxygen removed through H_2O = 11.332 kgmoles O
Oxygen generated = 38.124 kgmoles - 11.332kgmoles = 26.792 kgmoles O
= 13.396 kgmoles O_2

$$13.396 \text{ kgmoles } O_2 * \frac{32 \text{ kg } O_2}{\text{kgmole}} = 428.7 \text{ kg } O_2 \text{ generated}$$

Step 8. Oxygen

The oxygen generated during a work tour was calculated in the previous step. To estimate how much the astronauts need, we go back to the food decomposition equation:

$$185 \text{ kg dry food} * \frac{1 \text{ kgmole food}}{26.72 \text{ kg food}} * \frac{1.2925 \text{ kgmole } O_2}{1 \text{ kgmole food}} * \frac{32 \text{ kg } O_2}{\text{kgmole}} = 286.4 \text{ kg } O_2$$

Oxygen generated	428.7 kg
Oxygen for respiration	286.4 kg
Module Leakage	42.0 kg
Excess Oxygen	100.3 kg

So there is no need to resupply oxygen.

Appendix D: Solar Array and Battery Sizing Calculations

These solar array sizing algorithms are contained in the Spacecraft Subsystems manual (Lozano, pp. 6-7):

Sample calculations for sizing a solar array for:

- 62 kW (end of life or EOL) power
- 10 years
- 10% electrical line losses
- 27 Nickel-Hydrogen batteries

Assume:

<u>Solar-array characteristics</u>

Silicon solar cells, efficiency = 12%
Individual cell size = 8cm x 8cm
Packing factor = 95%
Operating temperature(worst case) = 67 °C
Temperature coefficient = -.5% per °C
Sun-angle(worst case) = 23.5°
Solar intensity at 1 A.U. = 1358 W/m^2
Life-time degradation in efficiency = 25%
Specific power (W/kg) = 30 Watts/kg

<u>Nickel-Hydrogen battery characteristics</u>

27 batteries connected in parallel
Battery capacity = 100 Amp-hours
Battery voltage = 28 Volts
Charge time = 1 hour (60 minutes of daylight per orbit)

1.) Calculate array voltage (array voltage must be greater than battery voltage to ensure a potential gradient for charging. 20% greater than battery voltage is standard.)

array voltage = 28V x 1.2 = **33.6V**

2.) Calculate required end-of-life (EOL) power

$$\text{Power(EOL)} = \text{(power for loads)x(losses) + power to charge batteries}$$
$$= (62{,}000\text{W})(1.1) + \frac{(100\ \text{Amp-hrs})(33.6\text{V})(27\ \text{batts.})}{1\ \text{hour}}$$
$$= \mathbf{158{,}920W}$$

3.) Calculate temperature effect

Temp. effect = (67 - 28) x .005 = **.195**

4.) Calculate beginning-of-life (BOL) required power

$$\text{Power (BOL)} = \frac{\text{Power(EOL)}}{\text{(degrade)x(cos. sun angle)x(temp.effect)}}$$
$$= \frac{158920\text{W}}{(1-.25)(\cos 23.5°)(1-.195)}$$
$$= \mathbf{287{,}027.6W}$$

5.) Calculate total cell area

$$\text{Total cell area} = \frac{\text{Power(BOL)}}{\text{(solar intensity)(efficiency)}}$$
$$= \frac{287027\text{W}}{(1358\text{W/m}^2)(0.12)}$$
$$= \mathbf{1761.34\ m^2}$$

6.) Calculate number of cells required

$$\text{\# cells} = \frac{\text{total cell area}}{\text{cell size}}$$

$$= \frac{(1761.34\ \text{m}^2)(10{,}000\ \text{c. factor})}{64}$$

$$= \mathbf{275{,}209\ silicon\ cells}$$

7.) Calculate array size

$$\text{Array size} = \frac{\text{total cell area}}{\text{packing factor}}$$

$$= (1761.34\ \text{m}^2)/(.95)$$

$$= \mathbf{1854\ m^2}$$

$$= \mathbf{19{,}946\ ft^2}$$

$$= \mathbf{232\ m^2/panel}$$

8.) Calculate array mass

$$\text{Array mass} = \frac{\text{power requirement of arrays}}{\text{specific power}}$$

$$= \frac{68000\text{W}}{30\frac{\text{W}}{\text{kg}}}$$

$$= \mathbf{2267\ kg}$$

Battery Sizing Calculations

These calculations follow the McDermott algorithm contained in Wertz and Larson, pp. 363-364. Sample calculations for sizing the number and mass of Ni-H_2 batteries required to supply 62 kW of power for 30 minutes of eclipse time.

The following equation can be used:

$$C_r = \frac{P_e T_e}{C_d N V_d n}$$

where:

C_r = Rated battery capacity (ampere-hours) = 100 Amp-hours
P_e = Average eclipse load (watts) = 62,000 W
T_e = Maximum eclipse time (hours) = .533 hours
C_d = limit on battery's depth of discharge = 50%
N = number of batteries = unknown
V_d = Battery's average discharge voltage (bus voltage) = 28V
n = Transmission efficiency between battery and load = 90%

Using these values in the equation gives a result of:

N = 27 Ni-H_2 batteries

A 50% depth of discharge extrapolates to a lifetime of approximately 2 years (Wertz and Larson, p. 363).

<u>Battery Mass Estimate</u>

The power or watt-hour capacity of these batteries is:

Watt-hour capacity per battery = (100 A-hrs) x (28V) = 2800 W-hrs

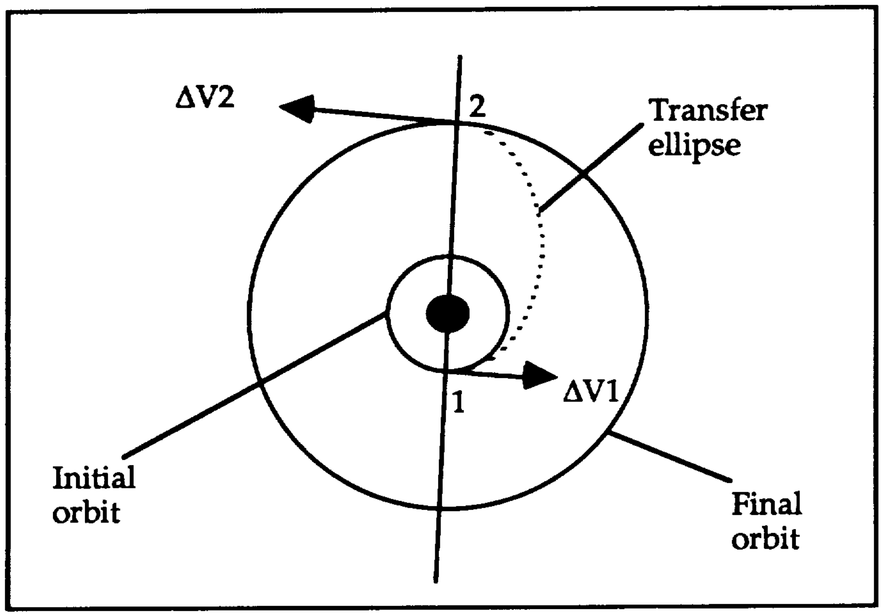
Using an energy density value of 25 W-hr/kg, the mass of each battery is obtained by simply dividing this value into the W-hr capacity:

$$\text{Mass} = \frac{2800 \text{ W-hrs}}{25 \frac{\text{W-hrs}}{\text{kg}}} = 112 \text{ kg per battery} = \mathbf{3024 \text{ kg}}$$

Appendix E: Reboost Calculations

ΔV Calculation:

Reboost (Hohman transfer):



Circular velocity at 1:

$$(V_{cir})_1 = \sqrt{\frac{\mu}{r_1}}$$

where

- μ universal gravitational constant (398600.44 km^3/sec^2)
- r_1 initial orbit radius.

Circular velocity at 2:

$$(V_{cir})_2 = \sqrt{\frac{\mu}{r_2}}$$

where

- r_2 final orbit radius.

Velocity at perigee (1) of transfer ellipse:

$$V_p = \sqrt{\mu\left(\frac{2}{r_1} - \frac{1}{a}\right)}$$

where

a semi-major axis of transfer ellipse $= \frac{r_1 + r_2}{a}$

Velocity at apogee (2) of transfer ellipse:

$$V_a = \sqrt{\mu\left(\frac{2}{r_2} - \frac{1}{a}\right)}$$

The velocity change required at perigee is:

$$\Delta V_1 = | V_1 - V_p |$$

The velocity change required at apogee is:

$$\Delta V_2 = | V_2 - V_a |$$

Total reboost ΔV(km/sec) required:

$$\Delta V_{total} = \Delta V_1 + \Delta V_2$$

$$\Delta V_{tot} = \sqrt{\mu}\left[\left| \sqrt{\left(\frac{2}{r_1} - \frac{1}{a}\right)} - \sqrt{\frac{\mu}{r_1}} \right| + \left| \sqrt{\left(\frac{2}{r_2} - \frac{1}{a}\right)} - \sqrt{\frac{\mu}{r_2}} \right|\right]$$

For SHARC:

r_1 = 6742 km

r_2 = 6758 km

a = 6750 km

μ = 398600.44 km^3/sec^2

Therefore ΔV_{tot} = 9.107 m/sec

Propellant Budget:

Reboost:

$$M_p = M_f\left[e^{(\Delta V/I_{sp}g)} - 1\right]$$

where

M_p	propellant required for reboost
M_f	final mass of spacecraft
I_{sp}	specific impulse of the popellant
g	gravitational acceleration of Earth (9.81 m/sec^2).

For SHARC:

M_f = 1786000 kg

Attitude control during reboost:

$$M_{\Delta V} = \frac{K_v M_{s/c} \Delta V l_v \alpha_v}{L_t g I_{sp}}$$

where

K_v	control system's effectivity (>1, typically 2)
$M_{s/c}$	mass of spacecraft
ΔV	reboost velocity
l_v	distance from velocity control thuster to center of mass
α_v	angular offset of thrust vector from the center of mass in radians (0.002~0.01 rad)
L_t	lever arm of the control thruster
g	gravitational acceleration

For SHARC:

K_v = 2
$M_{s/c}$ = 1786000 kg
ΔV = 9.107 m/sec
l_v = 32.6 m
α_v = 0.005 rad
L_t = 30.8 m
$M_{\Delta V}$ = 159.5 kg

Attitude control during normal orbit:

$$M_{attman} = \frac{I_c \theta_m}{T L_t g I_{sp}}$$

where

I_c	moment of inertia about control axis
θ_m	rotation angle required (dead-zone size = $\pm\theta_m$)
T	time required to complete attitude maneuver
L_t	thruster lever arm about control axis.

External torque disturbances:

Drag:

$$F_d = \frac{1}{2}\rho C_d A V^2$$

where

ρ	density at given altitude
C_d	coefficient of drag
A	effective area perpendicular to direction of velocity
V	velocity of spacecraft.

The torque due to drag is given by the equation

$$T_d = F_d L_d$$

where

F_d	drag force acting on effective area perpendicular to velocity direction
L_d	distance between center of effective area and center of mass.

For SHARC:

ρ	$= 4.013 \times 10^{-12}$ kg/m^3
Cd	= 2.0
A	= 1354 m^2
L_d	= 27.8 m
T_d	$= 1.511 \times 10^{-7}$ N-m

Gravity gradient at orientation of 90 degrees from local vertical:

$$T_g = \frac{3\mu}{r^3} \left| I_z - I_y \right| \theta$$

where

r^3	radius of orbit
I_z	mass moment of inertia of spacecraft about z-axis (local vertical axis)
I_y	mass moment of inertia of spacecraft about y-axis(axis perpendicular to both local vertical axis and velocity axis)
θ	maximum deviation from local vertical (in radians).

For SHARC:

r	= 6742 km
I_z	= 668273955.4 kg-m^2
I_y	= 587705912.7 kg-m^2
θ	$= \pi/2$ radians
T_g	= 49.4 N-m

Appendix F: Thermal Control Calculations

F.1 Thermal Model Theory

The initial thermal estimate was a simple energy balance method outlined in Griffin and French. Three energy inputs were considered: direct solar radiation, reflected solar radiation, and internal energy generation.

The direct solar term was calculated using the blackbody asborption equations:

$$Q_{Sun} = (\alpha_1 A_1 + \alpha_2 A_2) I_{Sun} \tag{F.1}$$

The terms for the thermal calculations are explained in the TK model below. The reflected solar term was calculated using similiar principles, except that the area terms now refer to the area projected towards the Earth. The flux term is multiplied by the percentage of the Earth's surface below SHARC that is in the sunlight and the average albedo of the Earth:

$$Q_{Earth} = aA_{lit}(\alpha_1 A_1 + \alpha_2 A_2) I_{Sun} \tag{F.2}$$

Internal heat generation was set at 10 kW as a rough estimate. As mentioned in the main report, further analysis is needed in this area.

Two heat sinks were identified, energy radiated towards the Earth and energy radiated towards outer space. The difference between the two terms is the Earth blackbody radiation:

$$Q_{Earth} = \sigma(\varepsilon_1 A_1 + \varepsilon_2 A_2) T^4 - \sigma(\alpha_1 A_1 + \alpha_2 A_2) T_e^4 \tag{F.3}$$

$$Q_{Space} = \sigma(\varepsilon_1 A_1 + \varepsilon_2 A_2) T^4 \tag{F.4}$$

The trusses and modules are assumed to be polished aluminum, and the solar panels are assumed to be fused quartz silica. Both sets of thermal characteristics are taken from Wertz and Larson. The thermal characteristics of the Earth were taken from Zeilik and Gaustad.

After we determined an equilibrium temperature for SHARC, we estimated how much energy would be required to cool them down to 21°C, which NASA defines as operating temperature for the Freedom habitation modules. The process is almost identical to the previous one, except that the terms include the module surface area, and reflected energy towards SHARC.

F.2 Station Thermal Model

```
QSun = (Absorp1*ASun1+Absorp2*ASun2)*ISun
QReflect = Albedo*ALit*(Absorp1*AEarth1+Absorp2*AEarth2)*ISun
Theta = ThDeg*2*Pi()/360
QtoEarth  =  StBoltz*(Emiss1*AEarth1+Emiss2*AEarth2)*TSharc^4-
      StBoltz*(Absorp1*AEarth1+Absorp2*AEarth2)
QtoSpace = StBoltz*(Emiss1*ASpace1+Emiss2*ASpace2)*TSharc^4
QSun + QReflect + QInt = QtoEarth + QtoSpace
AEarth1 = 800*Cos(Theta)*Cos(Inc)
Inc = IncDeg*2*Pi()/360
ALit = 1 - 0.5*Sin(Theta)
Area1 = ASpace1+AEarth1
```

St	Input	Name	Output	Unit	Comment
		QSun	1005768		Solar Energy Input
		QReflec	182907.6		Solar Energy Reflected by Earth
	10000	QInt			Internal Energy Generation
		QtoEart	400582.86		Energy radiated towards Earth
		QtoSpac	798092.73		Energy radiated towards space
	.805	Absorp1			Solar Panel Absorptivity
	.825	Emiss1			Solar Panel Emissivity
	200	ASun1			Solar Panel Area facing Sun
		AEarth1	565.68542		Solar Panel Area facing Earth
		ASpace1	1034.3146		Solar Panel Area facing space
	1600	Area1			Total Solar Panel Area
	.2	Absorp2			Truss Absorptivity
	.031	Emiss2			Truss Emissivity
	380	ASun2			Truss Area facing Sun
	255	AEarth2			Truss Area facing Earth
	2945	ASpace2			Truss Area facing space
	3200	Area2			Total Truss Area
	5.67E-8	StBoltz			Stefan-Boltzmann Constant
	1396.9	ISun			Solar flux at LEO
	.4	Albedo			Earth Average Albedo

	ALit	.64644661	% of Earth Surface Reflecting
290	TEarth		Av. Blackbody Temp. of Earth
	Theta	.78539816	Orbital Position
	Inc	0	Orbit Inclination
45	ThDeg		Orbital Position in degrees
0	IncDeg		Orbit Inclination in degrees
	TSharc	349.38565	Average Temperature of SHARC

F.3 Module Thermal Model

```
QSun + QReflect + QInt = QtoEarth + QtoSHARC + QtoSpace
QSun = Absorp*ASun*ISun
QReflect = Albedo*Absorp*ALit*AEarth*ISun
QtoEarth = StBoltz*AEarth*(Emiss*TMod^4-Absorp*TEarth^4)
QtoSHARC = StBoltz*ASharc*Absorp*(Emiss*(1-AbSharc)*
      TMod^4-ESharc*TSharc^4)
QtoSpace = StBoltz*ASpace*Emiss*TMod^4
ThermPow = QInt - 4028
```

St	Input	Name	Output	Unit	Comment
		QSun	29832.196		Solar energy input
		QReflec	6225.7039		Solar energy reflected by Earth
		QInt	-42106.96		Internal energy generation
		QtoEart	-7473.337		Energy radiated towards Earth
		QtoSpac	2398.5775		Energy radiated towards space
		QtoSHAR	-974.3049		Energy radiated towards SHARC
	.2	Absorp			Module absorptivity
	.031	Emiss			Module Emissivity
	.379	AbSharc			Absorptivity of SHARC
	.0346	ESharc			Emissivity of SHARC
	106.78	ASun			Module area facing sun
	182.65	ASpace			Module area facing space
	182.65	ASharc			Module area facing SHARC
	111.42	AEarth			Module area facing Earth
	294	TMod			Average temperature of module
	290	TEarth			Average blackbody temp of Earth
	365.01637	TSharc			Average tempeature of SHARC
	1396.9	ISun			Solar energy flux at LEO
	.4	Albedo			Average albedo of Earth
	.5	ALit			% of Earth's surface reflecting
	5.67E-8	StBoltz			Stefan-Boltzmann constant
		ThermPo	-46134.96		Power req. for Thermal Control

(Note that a negative value for ThermPow indicates the amount of heat that must be dissipated)

Appendix G: Mass Calculations

Power	Quantity	Mass (kg)	Total
Solar Panels	8	258	2067
Regulator and Converter	1	2268	2268
Batteries	27	3024	81648
Control Unit	1	1240	1240
Lights	20	1	20
Wire	364 ft x .25 in dia	150	150
	Total		87391

Crew and Life Support	Quantity	Mass (kg)	Total
Habitation Module	2	16329	32658
Control Module	2	16329	32658
40' x 20' Pressurized Vessel	1	16329	16329
Pressure tubes and airlocks	2 of each	8165	16329
Closed Loop Hardware	1	782	782
Escape Pod	1	23140	23140
Consumables	1 two month period	489	489
	Total		122386

Propulsion	Quantity	Mass (kg)	Total
Attitude Thrusters	20	100	2000
Reboost Thrusters	2	150	300
Propellant N_2O_4/MMH	2	4870	9740
Propellant Hydrazine	2	1100	2200
Storage Tanks	24	94	2256
	Total		16496

Truss Structure	Quantity	Mass (kg)	Total
Double Fold Deployable	2	18598	37196
Single Deployable	1	3266	3266
Erectable	4	828	3312
Robotic Track	520 m	5000	5000
	Total		48774

Robotics	Quantity	Mass (kg)	Total
Large Arm	3	1000	3000
Small Arm	4	1000	4000
	Total		7000

Thermal Control	Quantity	Mass (kg)	Total
Radiator	1	4512	4512
	Total		4512

Communications	Quantity	Mass (kg)	Total
Communication package	1	200	200
	Total		200

SHARC's Total Mass = 287005 kg